NASA

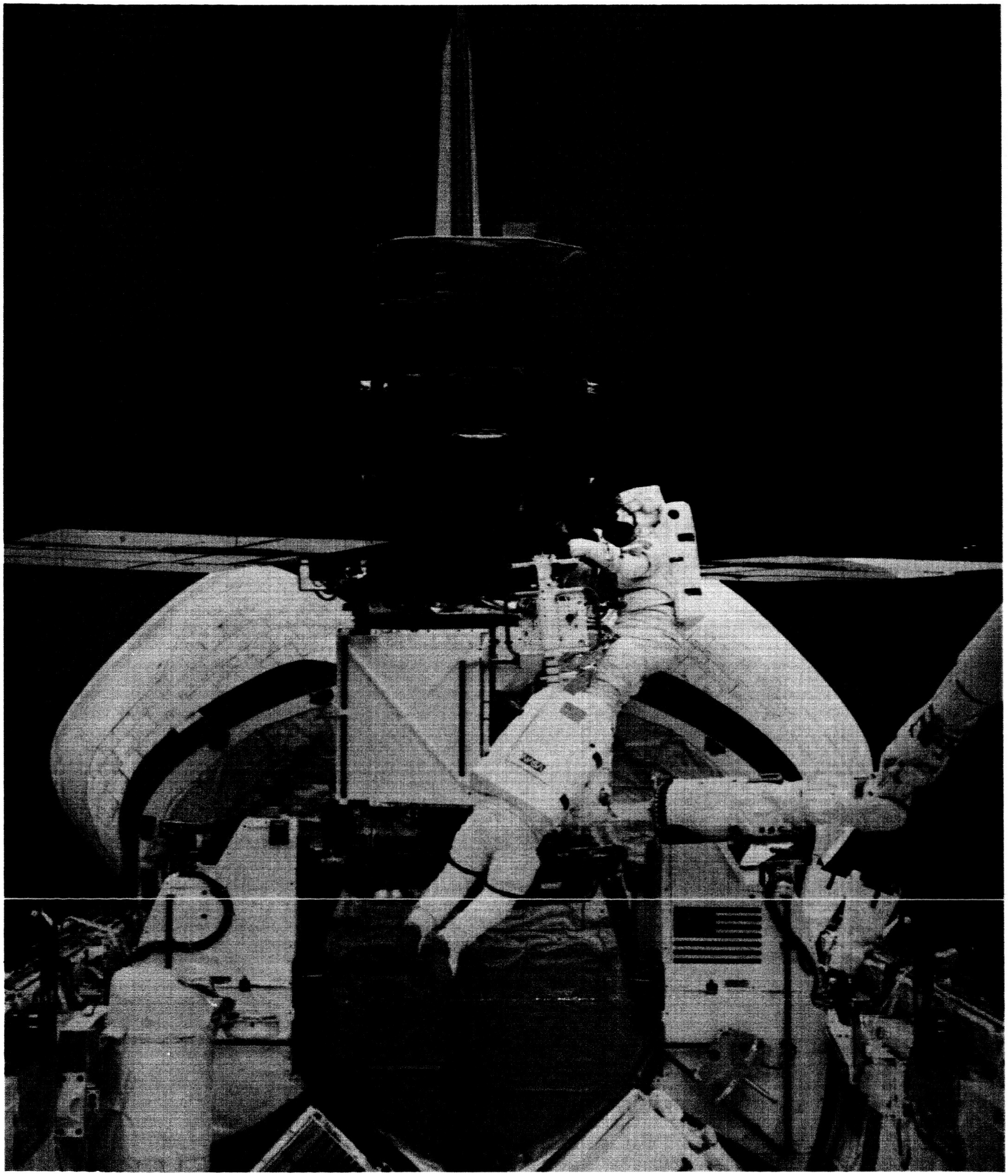
National Aeronautics and Space Administration

Lyndon B. Johnson Space Center
Houston, Texas 77058

Satellite Services Workshop II

November 6-8, 1985

Volume 1



JSC-20677

SATELLITE SERVICES WORKSHOP II

November 6-8, 1985

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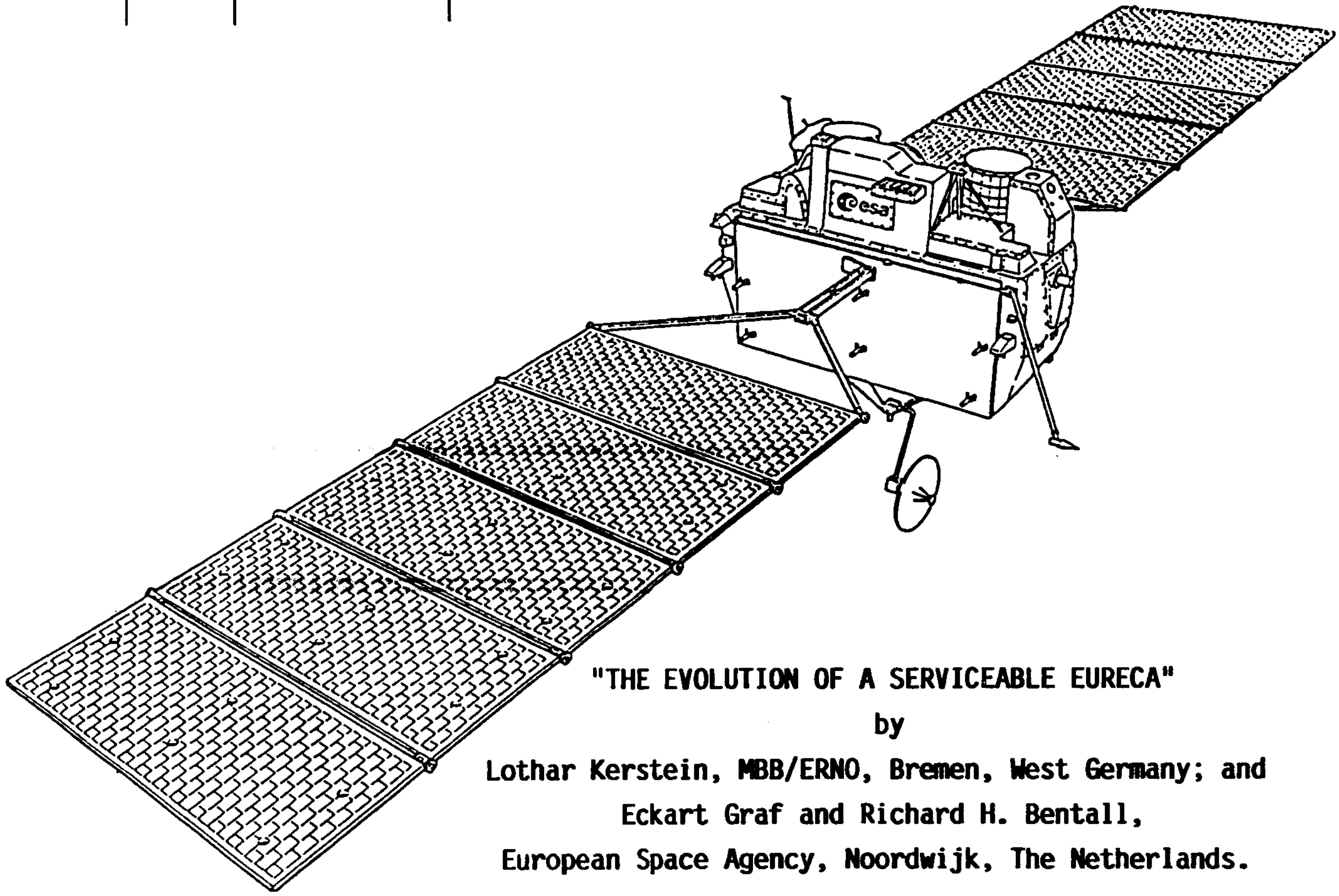

"THE EVOLUTION OF A SERVICEABLE EURECA"

by

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Abstract of the Paper:

EURECA is now becoming established as a ground serviced platform for experimental packages. As the facilities available for in-orbit servicing improve, culminating in the Space Station Programme, there is a growing interest in evolving the EURECA platform into a serviceable spacecraft, allowing the exchange of payloads in orbit, and the achievement of longer missions. This paper describes the expected serviceable features of an EURECA and describes a demonstration mission approach which would constitute an early and important stage of the development towards a fully space-based platform.

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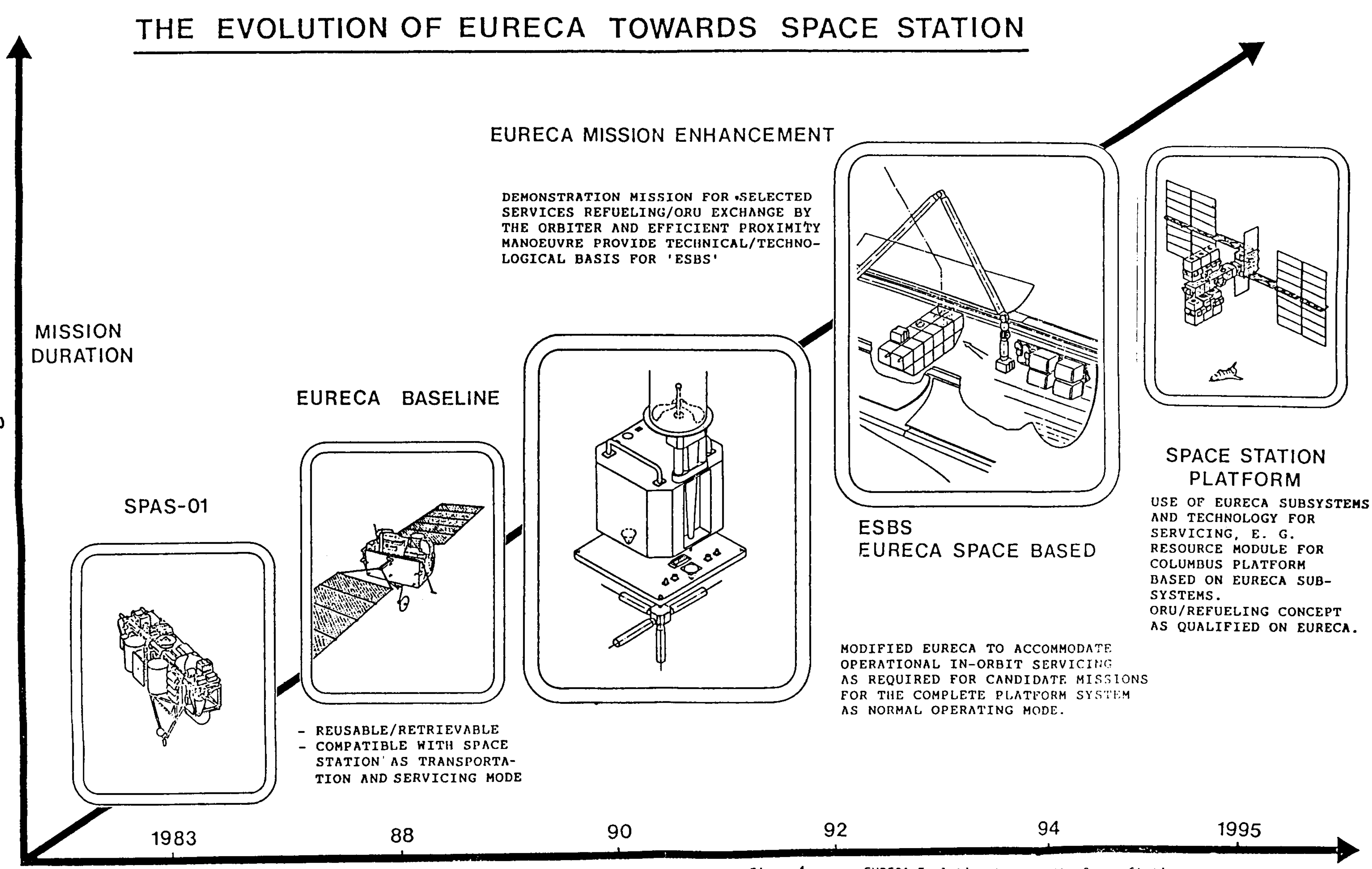


Fig. 1: EURECA Evolution towards the Space Station

DESCRIPTION OF THE BASELINE EURECA

PROGRAMME SUMMARY

The European Retrievable Carrier (EURECA) is a free-flying reusable platform launched and retrieved by the STS. As an element of the Spacelab follow-on development programme EURECA provides to the user community a platform with capabilities beyond those of Spacelab regarding on-orbit staytime and microgravity environment and will allow important research and application missions prior to as well as complementary to the Space Station for payloads which do not require man's involvement.

While the first EURECA mission will be primarily a microgravity mission, the cost effectiveness of reutilisation of an available retrievable platform is also of interest for the space science community, particularly astronomy and solar physics, and allows flight opportunities for a variety of earth observation payloads. In addition, EURECA constitutes an ideal test bed for in-orbit demonstration of technologies like inter-orbit communication, rendez-vous and docking, and in-orbit servicing, which are essential for Europe to achieve its long-term objectives in space.

Consistent with the initial objectives of the Eureca programme to expand Europe's capability and competitiveness in the development, utilisation, and operation of low earth orbiting platforms. Eureca also provides the essential basis for technologies and operational capabilities required for several candidate elements within the European space station scenario.

Several important features of the baseline Eureca design are directly applicable to its utilisation as a co-orbiting and non co-orbiting space station platform. These are planned for demonstration and qualification during the first mission and will include orbit change capability, rendez-vous with a target point in orbit in support of retrieval by the orbiter, activation/deactivation of Eureca including safety critical operations in orbiter proximity, European mission and payload control, ground operations and logistics for retrievable, reuseable platforms. Further it is planned to demonstrate platform/European Data Relay Satellite/ground communication capability, using Eureca in combination with the Olympus satellite.

EURECA constitutes the nucleus of a resource module, the performance of which can be adapted to cover evolving user requirements in a smooth and low-cost programme evolution. The capability of in-orbit servicing of subsystems and payloads can be implemented gradually in correct phasing with realistic European mission requirements and evolving space station architectures, interfaces and economics.

EURECA is an approved programme of the European Space Agency. The phase C/D started in December 1984. Planned launch dates for the first mission are March 1988 and September 1988 for deployment and retrieval, respectively.

The overall EURECA concept and programme objectives are summarized as follows:

THE OVERALL CONCEPT

Frequent Flight Opportunities at Low Cost.

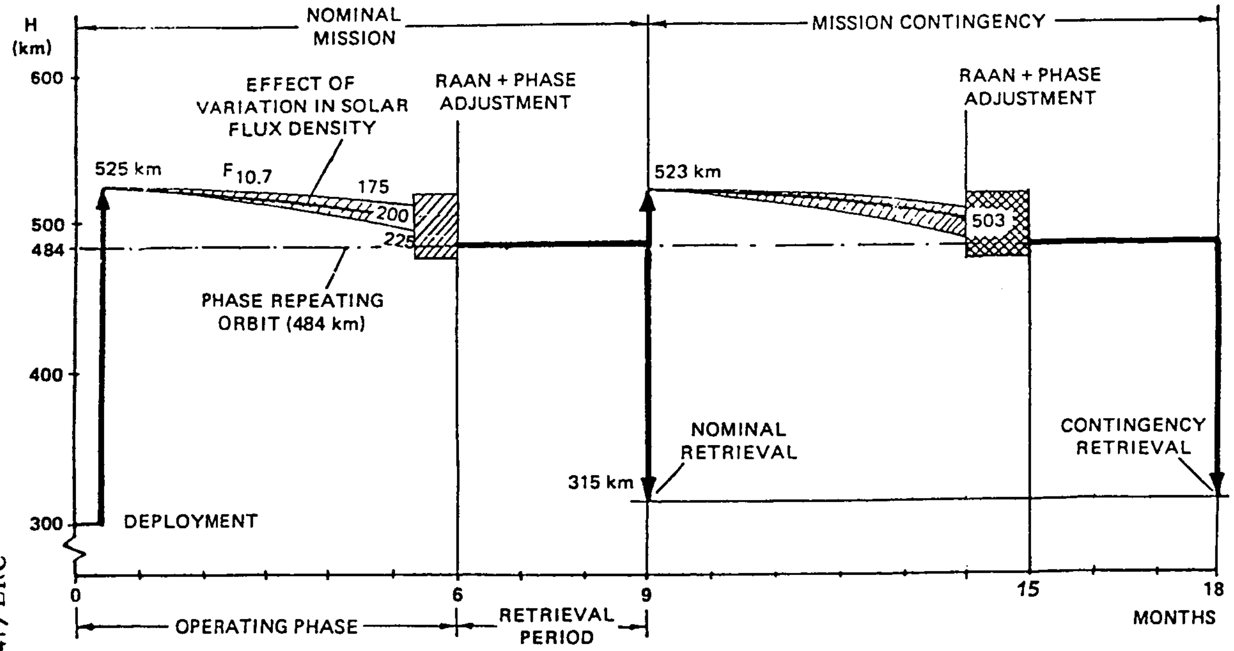
- o Retrievable, reusable platform
- o Six to nine months operation
- o five missions or ten years total lifetime
- o Standardized payload interfaces, integration and check out
- o Low cost ground and flight operations
- o Short turnaround o Low transportation cost

PROGRAMME OBJECTIVES

- o Offer frequent flight opportunities at low cost
- o Meet known platform user requirements for microgravity, space science, earth observation, technology
- o Establish a concept of retrievable, reusable platforms which can be adapted to meet evolving mission requirements
- o Develop European capabilities in space platform design, development, utilization and operation
- o Develop an initial platform which meets essential design, operational and programmatic requirements of future space station elements.

The long life-time capability of the EURECA spacecraft systems contrasts with the short duration of the experiments and instruments which are its customers, and it is therefore a great interest to maintain the Eureca in orbit, while exchanging only those payloads which have served their purpose.

Fig. 2 THE MISSION PROFILE



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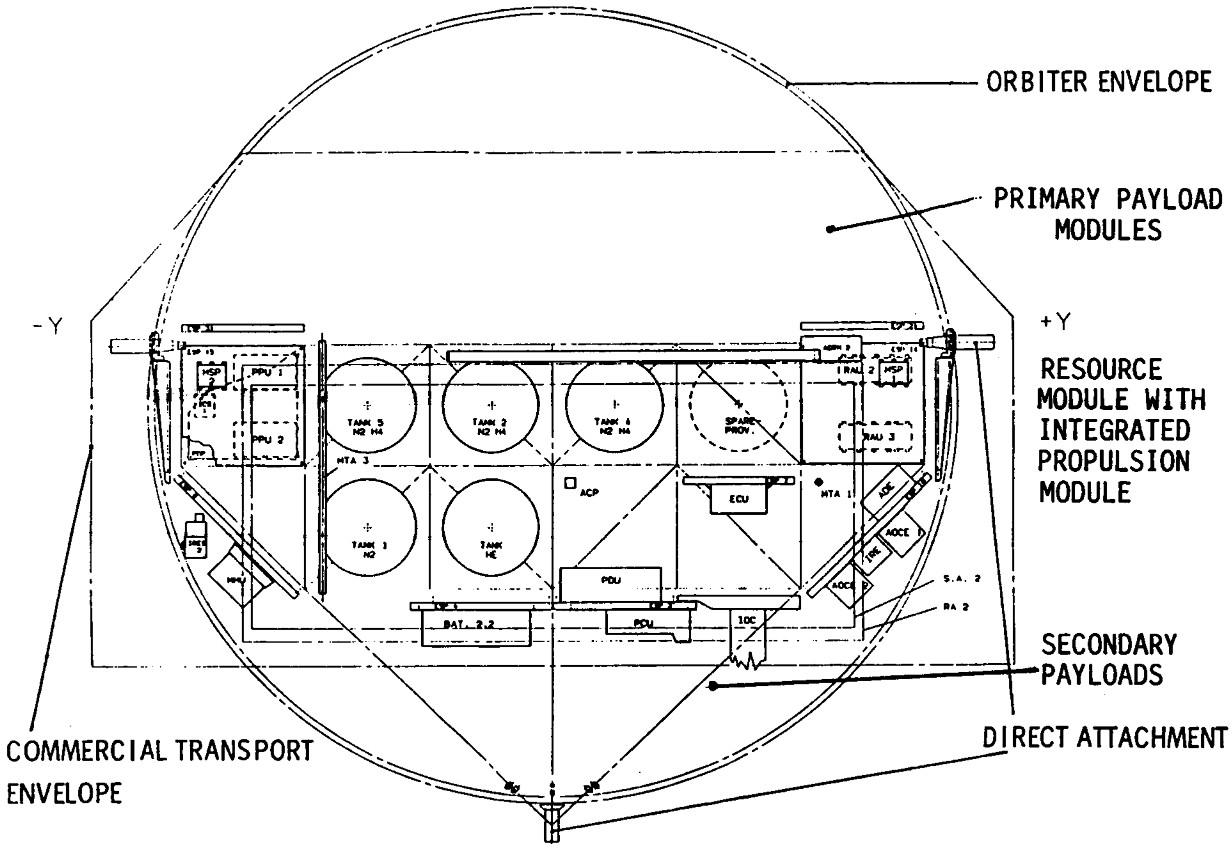
EURECA

Fig. 3: **THE CONFIGURATION**



Tab. 1 **EURECA SYSTEM CAPABILITIES**

MASS : TOTAL : 4000 kg
AVAILABLE TO PAYLOAD : 1000 kg

VOLUME : AVAILABLE TO PAYLOAD : 8,5 m^3

POWER : AVAILABLE TO PAYLOAD : 1000 W
PEAK. : 1500 W
SOLAR ARRAY OUTPUT : 5000 W

THERMAL CONTROL : LIQUID FREON LOOP (1000 W) AND MULTI LAYER INSULATION

DATA MANAGEMENT : HIGH SPEED : 256 kbps
LOW SPEED : 2 kbps
MEMORY CAPACITY : 128 Mbits
AVERAGE P/L : 1,5 kbps

ATTITUDE POINTING ACCURACY : ± 1° (3 SIGMA)

MICROGRAVITY : 10^{-5} g< 1 Hz
10^{-3} g> 100 Hz

ORBIT : 525 km ; 28.5°

MISSION DURATION : 6 MONTHS OPERATIONAL + 3 MONTHS
DESIGN LIFE : 5 MISSIONS OR 10 YEARS

TURN AROUND TIME :
BASELINE : <1.5 YEARS BETWEEN RETRIEVAL AND NEXT LAUNCH. REDUCTION DOWN TO LESS THAN ONE YEAR UNDER STUDY

EURECA/ORBITER INTERFACES

- STANDARD 3-POINT STRUCTURAL ATTACHMENT
- DEPLOYMENT/RETRIEVAL WITH REMOTE MANIPULATOR
- REMOTELY REMATABLE UMBILICAL FOR POWER AND DATA
- GRAPPLE FIXTURE/RMS ELECTRICAL INTERFACE FOR INITIAL ACTIVATION/FINAL DEACTIVATION AND AS BACK-UP FOR UMBILICAL POWER

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EURECA AND THE SPACE STATION

The decision in January 1985 to collaborate with NASA in the development of a manned Space Station is probably one of the most far-reaching ever made by ESA's member states. It is a decision which evokes strong feelings within the space community of Europe, ranging from cautions to unbridled enthusiasm.

In 1988, EURECA will be a proven system, designed expressly for retrieval and refurbishment. It will also be capable of adaptation to the tasks of carrying payloads pertaining to the major scientific and technological disciplines.

In addition, EURECA is also a candidate for part of the eventual Space Station infrastructure as an autonomous payload carrier. As such, it could accommodate and operate space instruments, either in the free-flying mode, the docked mode, or even tethered to the Space Station or to one of the free-flying platforms. The means to access the platform and at the same time maintain its operational capability have not yet been identified. An advanced EURECA platform may provide the answer.

The early availability of EURECA and its present shuttle-compatible features enable its use as an "Orbital Test-Bed" for the demonstration of technologies and techniques of potential international interest.

- o optimized mission profiles to minimize propellant consumption and/or to extend the on-orbit staytime,
- o proximity operations in support of retrieval by the Orbiter,
- demonstration of Orbital Replaceable Unit exchange,
- demonstration of refuelling.

The later two demonstrations (Refuelling and ORU exchange) with EURECA will be described in this paper. In order to allow a combined in-orbit demonstration for Refuelling and ORU exchange, the EURECA modifications would be implemented after the first mission in 1988, so that the in-orbit demonstration programme can be performed in 1990, early enough to benefit the Columbus application. The objective of this paper is also to assess the design improvements and evolutionary steps needed, so that the Space Station (COLUMBUS) program may make maximum and effective use of this platform (refer also to Fig. 1, EURECA Evolution towards the Space Station).

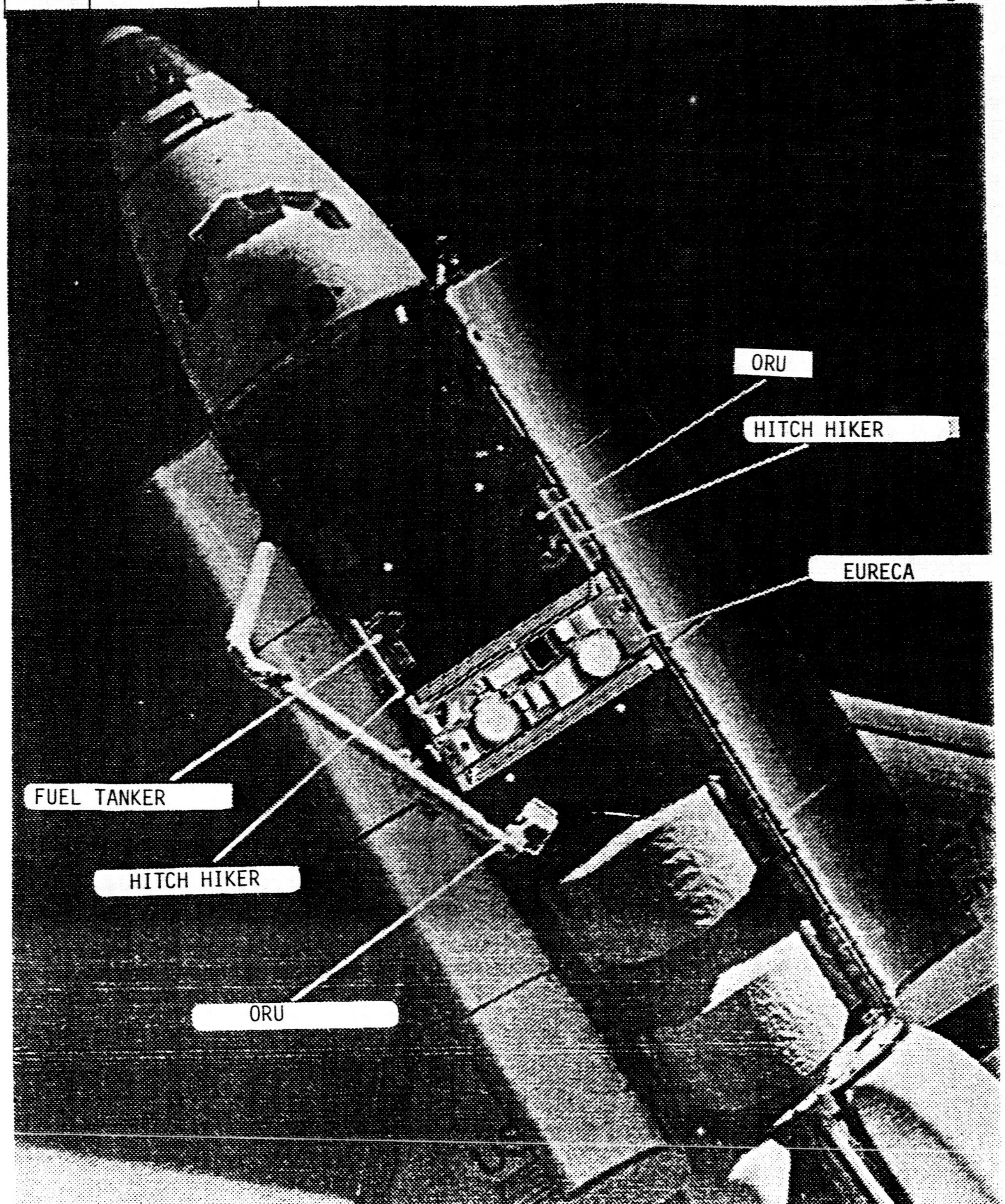


Fig. 4: EURECA as a TEST BED
A combined demonstration mission for refuelling and ORU exchange.

EURECA AS A TESTBED FOR IN-ORBIT DEMONSTRATION MISSIONS

IN-ORBIT REFUELLING

EURECA Orbit Transfer Assembly (OTA)

The task of the EURECA OTA is to transfer the carrier from its STS deployment altitude (= 300 km) to a higher orbit which is designated the operational orbit. This orbit is, depending on the OTA configuration (six-tank or eight-tank version), between 500 and 700 km. The OTA can fulfil transfer as well as attitude control tasks.

The OTA is a pressure-regulated monopropellant system, working with hydrazine as propellant and helium as pressurant. The control function is part of the Propellant and Pressurant Loading and Control Assembly (PPLCA). The baseline OTA configuration is shown in Figure 5.

600 kg (800 kg) hydrazine are stored in 6 (8) diaphragm tanks. These tanks are pressurized from one high pressure helium tank with a storage capacity of 4.6 kg helium, in which the operational temperatures range from 4 to 40° C, the max. operational pressure is 280 bar. A pressure of 23 - 24 bar within the propellant tanks is provided by two redundant pressure regulators. The propellant is distributed via a tubing system controlled by latching valves and pressure transducers into two redundant thruster branches consisting each of four 20 N thrusters. A thruster consists of a flow control valve and a thrust chamber assembly working with a catalyst that decomposes the injected hydrazine and expels the reaction products via the thruster nozzle.

EURECA REACTION CONTROL ASSEMBLY (RCA)

The RCA performs the attitude control of EURECA during its operational phase (μ-g mission) and during the deployment and retrieval phases in the vicinity of the STS, where a cold gas system is mandatory due to the safety requirements. The positioning of its twelve 20 mN thrusters provides for high control torques with large lever arms. They are arranged in two redundant branches, working at 1.5 bar operating pressure. Two redundant regulators, latching valves and high and low pressure transducers serve for control of the RCA. They are located on the Pressurant Distribution Panel (PDP). The 3 (5) RCA gas tanks are capable of storing 85 kg (142 kg) nitrogen. The operational temperatures range from -10 to +50 °C, the max. operational pressure is 280 bar.

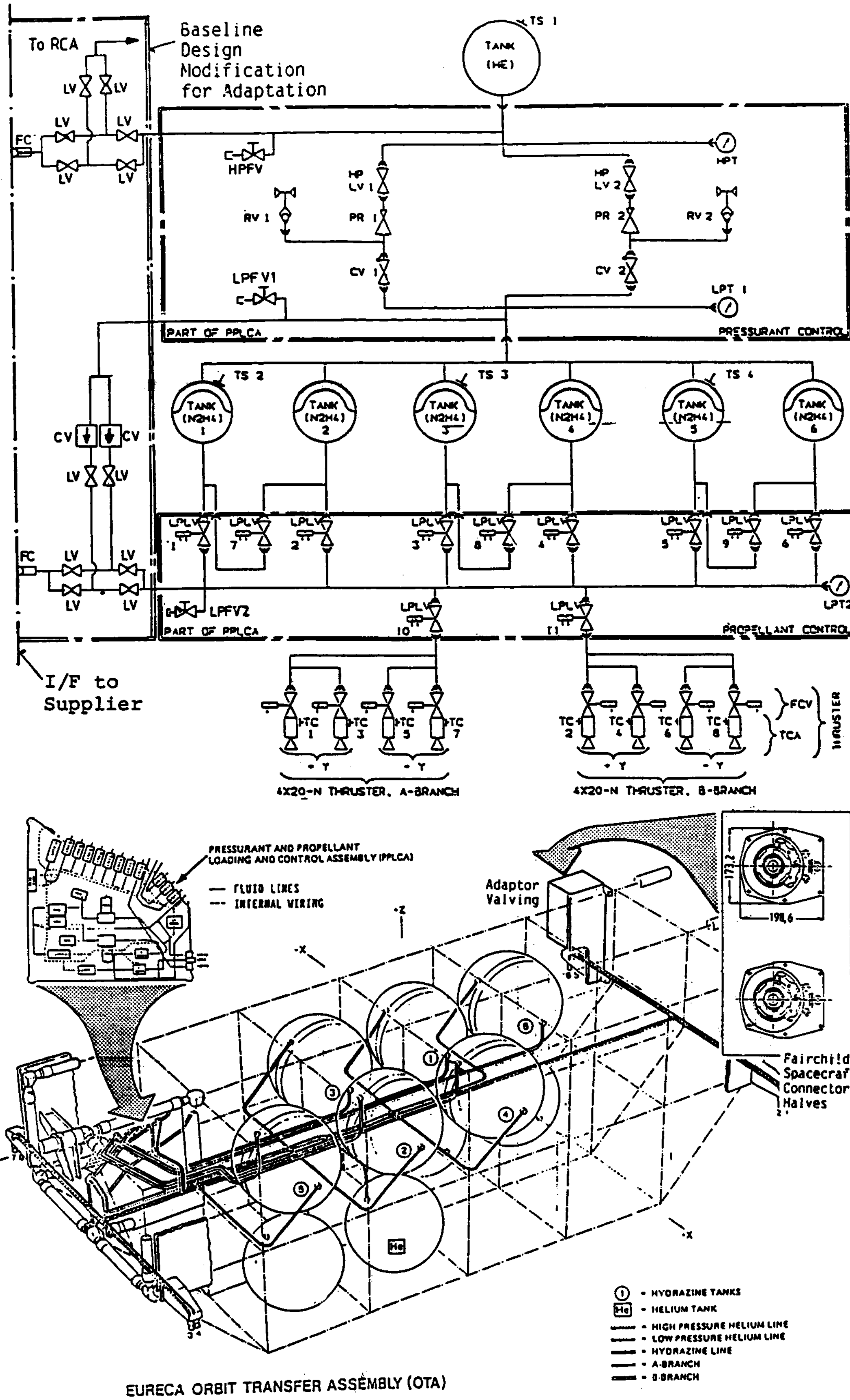


Fig. 5: The baseline OTA Configuration with modification for in-orbit refuelling

EURECA IN-ORBIT REFUELLING MODIFICATION ASPECTS

POSSIBLE REFUELLING CONCEPTS

To achieve a cost-effective refuelling demonstration, the method chosen should have minimum impact on the design. This rules out such methods as tank or module replacement and favours the utilisation of direct fluid transfer from a shuttle based (for the purposes of the experiment) servicing kit. Problems associated with the refuelling are typical also for the Space Station scenario, and include:

- contamination,
- safety,
- redundancy,
- temperature/pressure control,
- measurement of transfered fuel,
- purging,
- ullage gas processing (e.g. purification/ decomposition, venting, re-utilisation, storage), and various connector related requirements.

Signal and power will also need to be exchanged and monitored during the experiment.

LIQUID PROPELLANT TRANSFER BY PRESSURANT

The EURECA Orbit Transfer Assembly (OTA) is a pressure-regulated system with hydrazine (N_2H_4) as monopropellant. The transfer therefore has to be performed by regulated pressurization (Fig. 6).

Simultaneously to propellant refuelling the ullage gas has either to be restored in a large disposal tank on the supplier side or purified/decomposed and vented. Since the waste gas storage necessitates a disposal tank of total OTA diaphragm tanks volume, the gas venting modification seems to be preferable. Replenishing of the high pressure He-tank on the receiver side is then realized by gas transfer in the blow-down mode.

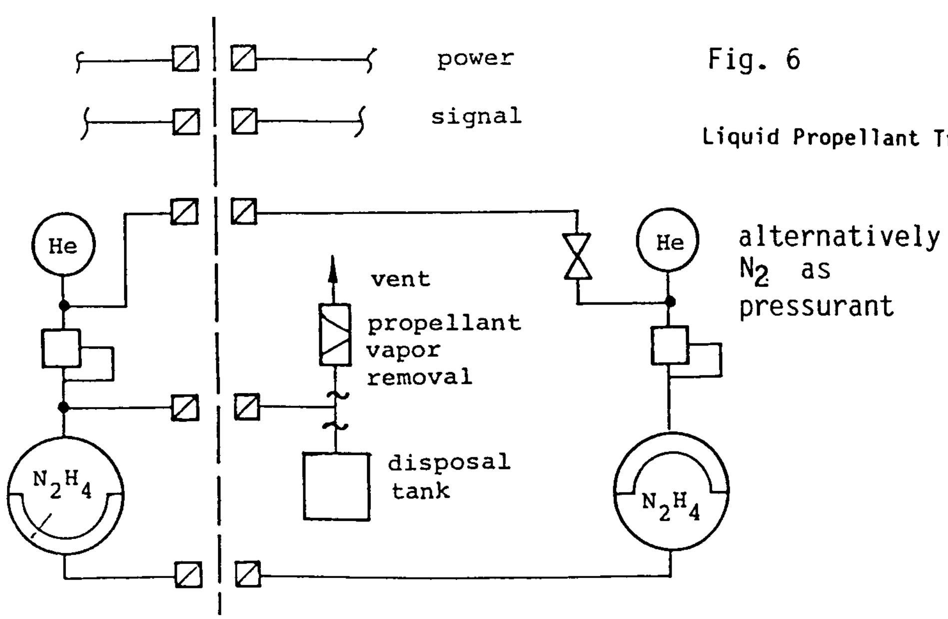


Fig. 6

Liquid Propellant Transfer by Pressurant

GASEOUS PROPELLANT TRANSFER BY BLOW-DOWN-MODE

The EURECA Reaction Control Assembly (RCA) uses GN_2 as cold gas propellant. Since N_2-tanks and refuelling pressures are equal to those of the He-pressurization system of OTA, the above mentioned replenishing concepts are applicable.

FLUID TRANSFER VIA FLEXIBLE HOSES/UMBILICAL

The proposed hoses/umbilical concept necessitates only a small modification or redesign of the EURECA configuration but requires EVA operations (example given in Figure 7).

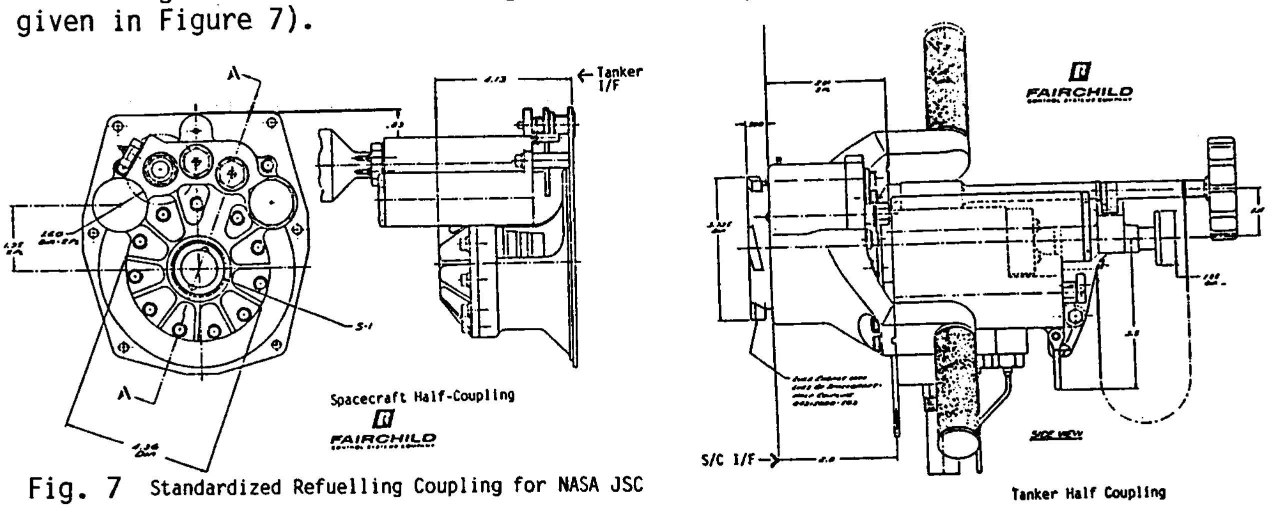


Fig. 7 Standardized Refuelling Coupling for NASA JSC

REFUELLING EXPERIMENT DESIGN

The detailed design tasks will be concentrated on those areas where critical refuelling configurations are identified during the conceptual phase of the in-orbit refuelling:

- o adaptation of EURECA baseline configuration for refuelling mission (relocation of fuel fill interfaces, etc.);
- o modification of EURECA Orbit Transfer Assembly (OTA hydrazine propellant and helium pressurant GSE connectors and interfaces, etc.);
- o design of fuel transfer adapter between Orbiter based refuelling system and EURECA OTA fluid/gas connectors;
- o modification of EURECA Reaction Control Assembly (RCA N_2 cold gas adapters, etc.) for pressurized gas transfer;
- o definition of the detailed refuelling sequence and procedure;
- o detailed design of electrical I/F adapter between EURECA GSE connectors and the refuelling system for power, command, and monitoring of pressure, temperature and valve positions;
- o incorporation of existing NASA hardware for in-orbit expendables resupply in the Orbiter cargo bay in the detailed design of the refuelling experiment, if advantageous.

ORU EXCHANGE DEMONSTRATION

ORU CONCEPT / BASIC CONSIDERATION

A concept of an Orbital Replaceable Unit has been established with the objective of performing an ORU demonstration mission with EURECA as a serviceable, small platform in LEO. The proposed ORU design and its implementation to the carrier effects only minor modification to the EURECA baseline.

The complete ORU design combines a suitable amount of existing technology and hardware with the results of European design studies. With regard to the ongoing COLUMBUS study, the design is also consistent with the interfaces to the US Space Station facilities as today indicated by the STS servicing capabilities.

One of the design objectives was to enable the verification of both functional and operational procedures with the demonstration model of the ORU, e.g. the concept of the EURECA operation and checkout will be changed by the usage of Orbit Replaceable Units (ORUs) (Fig. 8).

The capability of exchange instruments in orbit implies the transfer of operational activities from ground to orbit. Interface verification, S/W installation/updates and demonstrations, performance of qualification/acceptance tests, long-term flight verification programs have to be envisaged. On the other hand, the ORU design concept will be driven in part by the Operations & Checkout Requirements.

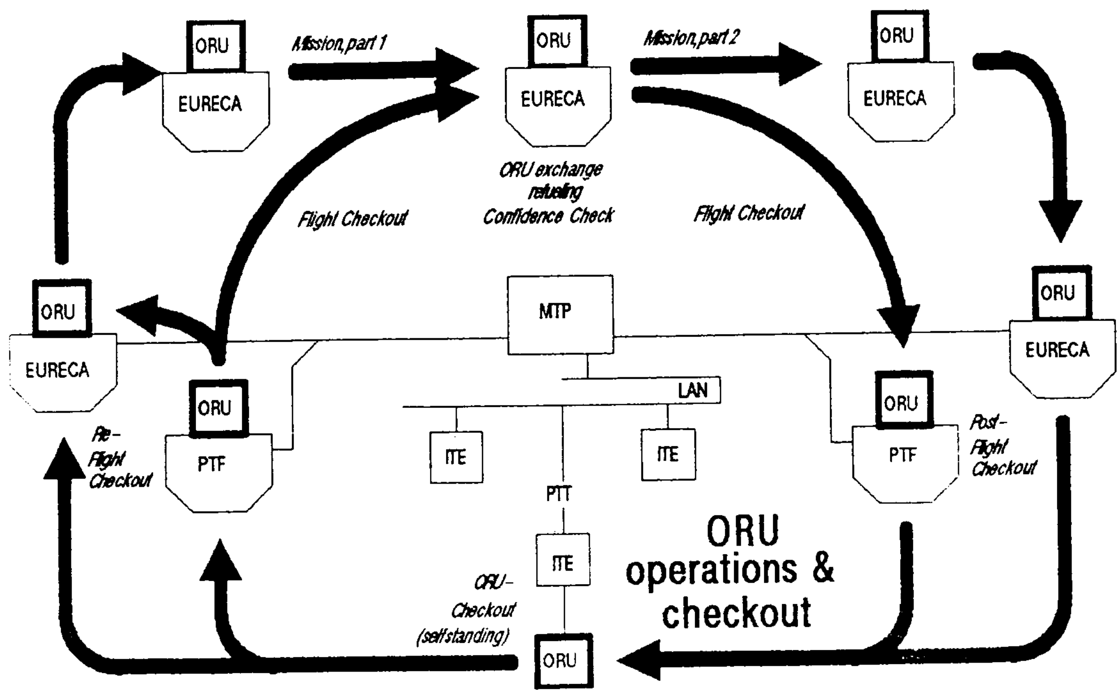
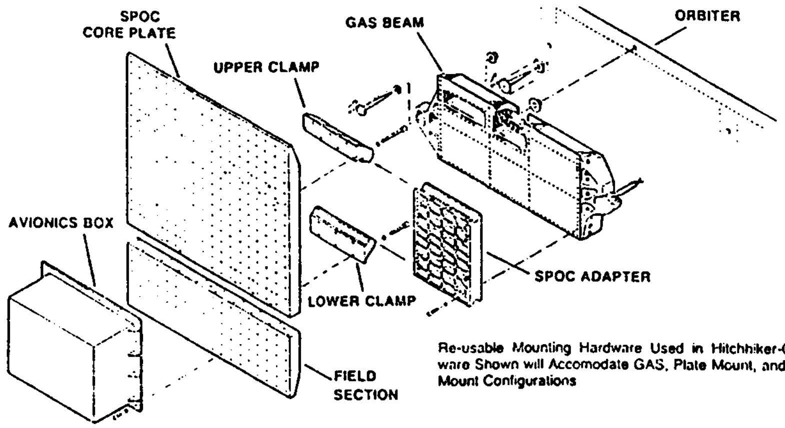


Fig. 8: Operations & Checkout Concept.

ORU DESIGN

The ORU design incorporates the following critical items:

- o cover structure and groundplate (see Fig. 10);
- o electrical connector H/W selection and integration;
- o ORU I/F to the servicing equipment (tool kit for bolt attachment i.e. UST , grapple fixture, etc.); the latching mechanism and the sequence of ORU attachment are shown in Fig. 10 also;
- o mechanical/electrical I/F between ORU and carrier structure;
- o definition of a carrier structure; this will include the NASA hitch-hiker-G program, the typical location and mounting I/F shows Fig. 9 for an ORU location during transport at the hitckhiker mounting interface;
- o ORU data handling I/F and performance requirements;
- o detailed planning and description of ORU replacement mission, using EVA operations.



Re-usable Mounting Hardware Used in Hitchhiker-G. The Hardware Shown will Accomodate GAS, Plate Mount, and Direct Mount Configurations

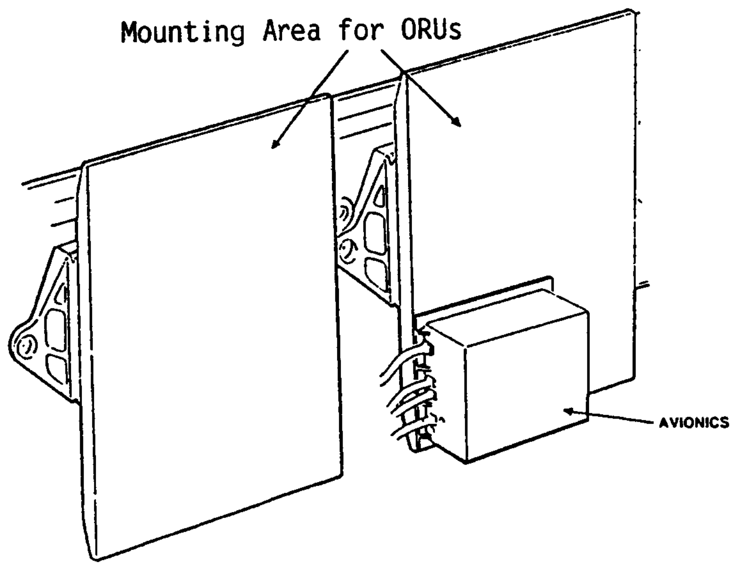


Fig. 9 : Proposed Location of ORU during Transport at STS Orbiter Hitchhiker Mouting I/F

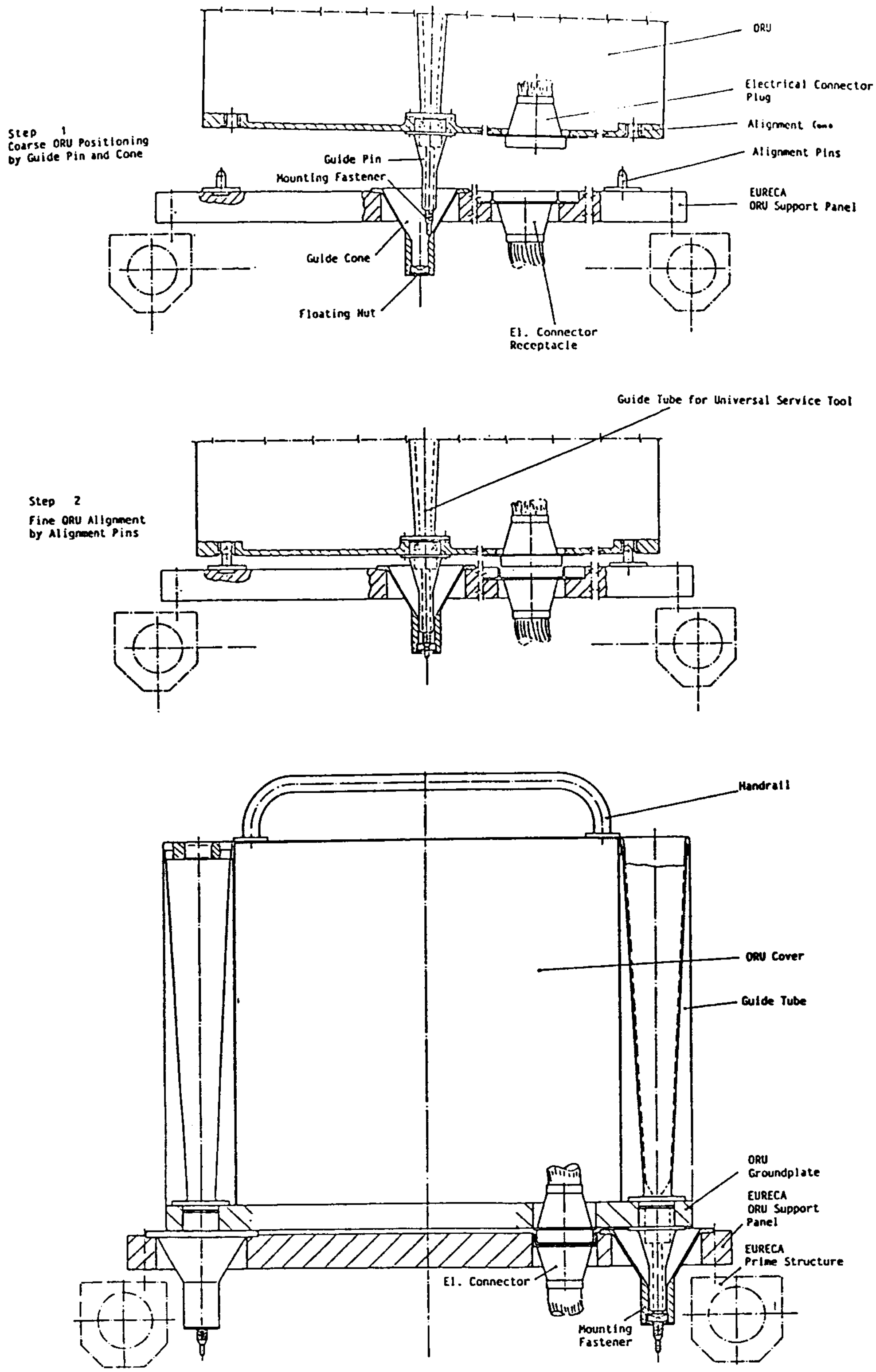


Fig. 10: ORU Exchange Sequence and Latching Mechanism

EURECA ENHANCEMENT FOR A SPACE-BASED APPLICATION

While the immediate objective of the programme is to utilise the Eureca as a test-bed for the flight proving of serviceable features, the ultimate goal is to evolve towards an "enhanced Eureca" whose payloads can be exchanged in orbit. Depending on mission and operational analyses for short turnaround missions, several or all instruments located on the upper platform can in principle be exchanged in orbit with EURECA berthed in the STS Orbiter cargo bay. The operational and detailed design concept of ORUs and the I/F to the shuttle servicing equipment is currently under investigation.

Based on MBB/ERNO's ORU design, an investigation has been performed to implement the payload ORU replacement as a fully integrated servicing function of EURECA, e.g. modification of the carrier and improvement of ORU concept. A possible concept of P/L ORUs attached to EURECA is shown in Figure 11.

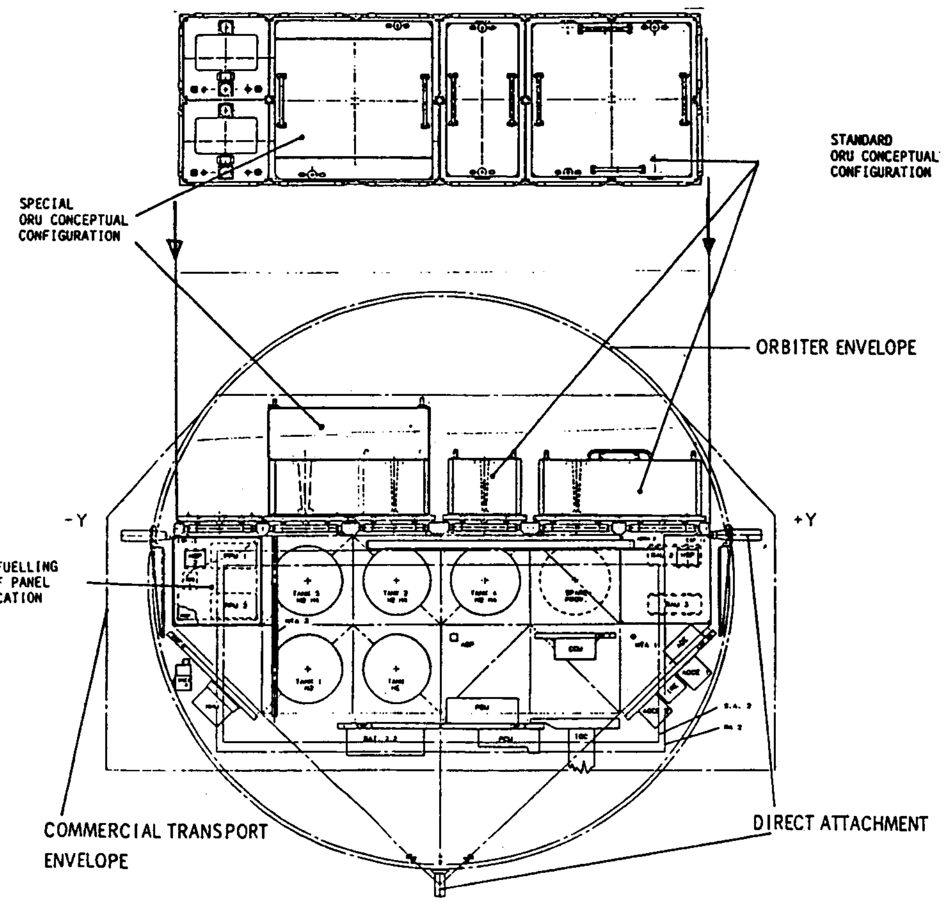


Fig. 11 Payload Orbital Replaceable Unit Concept.

For more complex applications, where technical improvements of the baseline EURECA are not feasible and cost effective, a new spacecraft concept is required. This can be most effectively achieved by a complete redesign, but using as much as possible existing EURECA and other European S/C elements and technology.

A two-module spacecraft will be studied (separate Resource Module (RM) and Payload Module (PM) with the objective to utilize especially for the RM the available EURECA or other European hardware. The application of conventional spacecraft design and technology with built-in redundancy will be applied in order to avoid highly sophisticated and over-emphasized modular design solutions. See Figure 12 "EURECA-Derived RM Concept".

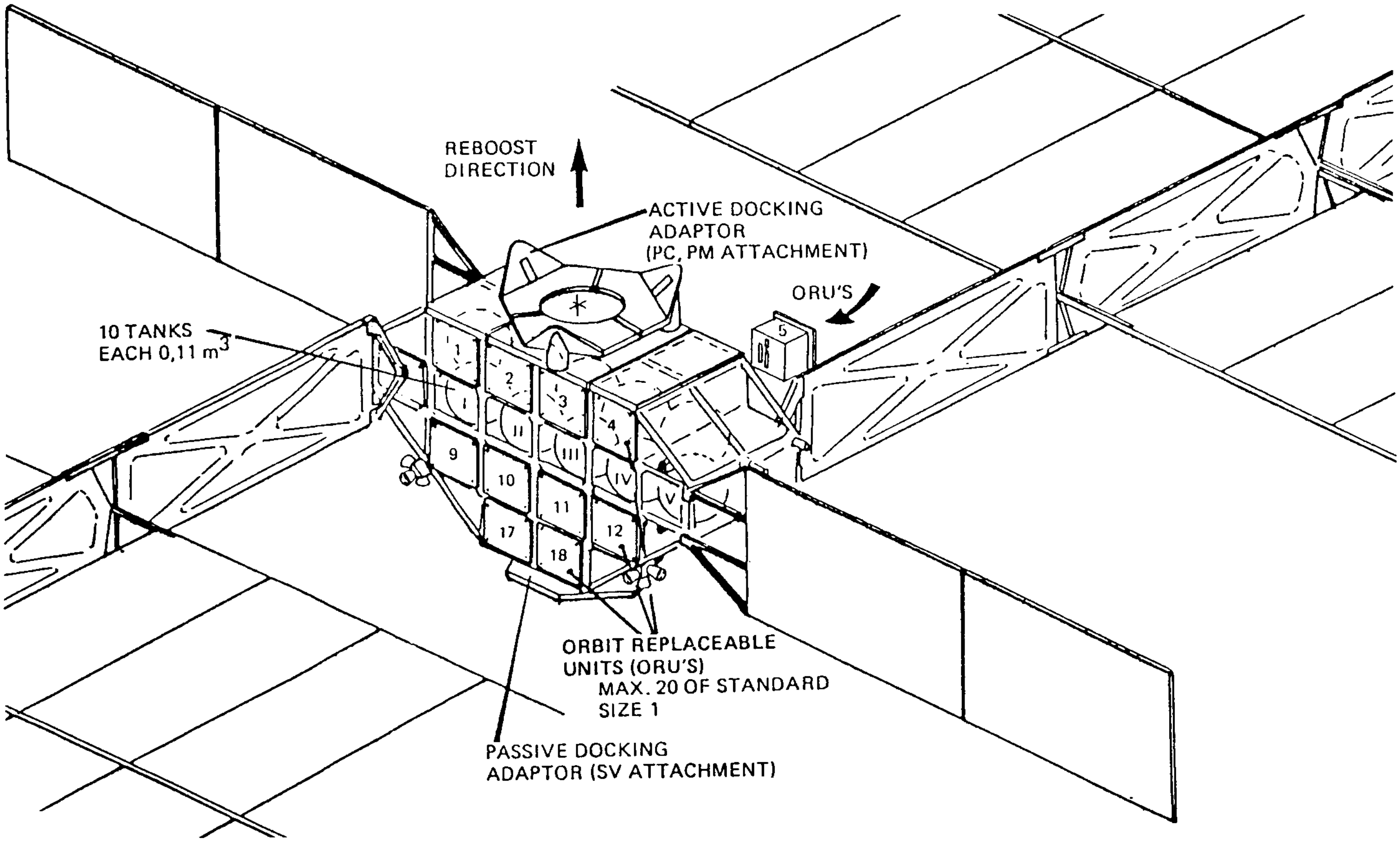


Figure 12 EURECA-Derived RM Concept

SUMMARY / CONCLUSION

This paper has described the design areas where current studies, intended to derive a serviceable Eureca, are concentrating their effort. The proposed demonstration encompasses aspects of servicing of interest not only to Europe, but also to the STS servicing capabilities and provides an early opportunity to achieve confidence in these important operational features.

SPACE STATION TECHNOLOGY EXPERIMENTS

by

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Satellite Services Workshop II
November 6-8, 1985
NASA Johnson Space Center
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SPACE STATION TECHNOLOGY EXPERIMENTS

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I. Abstract

With the advent of the Space Station will come significant opportunities to perform experiments in the near-Earth space environment. It is anticipated that the Space Station will be an in-space facility where long duration missions can be conducted. A large number of experiments are expected to be performed in science and applications, technology, and commercial ventures. NASA is working very actively to establish the experimental requirements from each experiment category. This paper addresses the in-space technology experiments and uses of the Space Station as presently envisioned for this next step in space.

II. Introduction

With the advent of the Space Station in the early 1990's will come opportunities to perform experiments in the near Earth space environment on a scale which far exceeds the current capability of the Space Shuttle or free fliers. The duration of the experiments will be on the order of weeks and months instead of several days as with the Shuttle. Also, it will be possible to conduct active experiments with man-in-the-loop to a far greater extent than presently possible. Since the Space Station will have associated platforms,[1] it will be possible to also perform in-space experiments which are isolated from the Station itself. This possibility permits experiments to be conducted at lower g levels than possible on the Space Station and at a different orbital inclination (90°) from the Station.

A large number of experiments are expected to be performed in science and applications, technology, and commercial ventures. Currently, the requirements for these classes of experiments are under assessment by NASA. This assessment is an extremely ambitious undertaking which involves more than the Space Station organizations at NASA Headquarters and the various NASA Centers. It also includes other NASA Headquarters program offices and significant resources from the NASA Centers. Technology experiments, the topic of interest in this paper, have been addressed by the Office of Space Station at NASA Headquarters. However, this activity has been primarily restricted to NASA conceptual experiments. Currently, the Office of Aeronautics and Space Technology, NASA Headquarters is undertaking an in-space research, technology, and engineering program to establish candidate activities for 1990 and beyond and to validate the associated experiment themes which best describe these activities.

III. Characteristics and Uses of the Space Station

Figure 1 shows the reference configuration of the Space Station, i.e., that configuration upon which the definition and design phase of the Space Station Program is based. The reference configuration is called the "Power Tower" configuration.[1] Its primary advantages are inherent stability (i.e., it will remain essentially Earth pointing without a concerted effort made to control its attitude); relatively unobstructed viewing angles for instruments and antennas; service accessibility for the Space Shuttle, the Orbital Maneuvering Vehicle (OMV), and satellites; and the facilitation of evolution or growth. The figure shows five modules for habitation, logistics, and laboratories arranged in a "race track" configuration. This module number and arrangement are under intense investigation, and both are subject to change by the time the design has been finalized. The figure shows a hoop-column structure attached to the Station which indicates the manner in which large space structures experiments can be accommodated as attached payloads on the Space Station. The large white boxes attached to the keel are for storage. There are also shown attached payloads at the top of the Station. The initial Station will be equipped with four sets of solar arrays and a pair of thermal radiators. An unmanned coorbiting platform is shown in the figure. This platform will be in the same orbital inclination as the Station but will be accessible from the Station by the Orbital Maneuvering Vehicle.

Figure 2 shows the purposes of the Space Station. It is intended to be a national laboratory in space where science and applications, technology, and commercial experiments can be performed. It will be a permanent observatory in low Earth orbit where the Earth and its environment and deep space can be remotely investigated. The Space Station will serve as a servicing facility for satellites, platforms and the OMV. It will serve as a node for the Space Transportation System. It will serve as an assembly facility and a manufacturing facility. The Space Station will serve as a storage depot to store expendables and experiments to be activated at a later date. It will also be a staging base for future programs in space. In every sense of the word a space station is a multi-purpose facility.

The uses of the initial Space Station are graphically depicted in figure 3. The three classes of experiments are shown with accompanying examples. Under science and applications, atmospheric and life sciences experiments will be conducted. The Earth Observing System[2] (EOS) is shown in the figure. This system will include an array of sensing instruments for investigating the Earth and its environment. Although the EOS is shown as a part of the main Space Station itself, in reality it may actually be accommodated on the polar (90° inclination) platform. Commercial experiments will include those which make use of a microgravity research facility. In the microgravity environment of such a facility it will be possible to produce pharmaceuticals and alloys that cannot be produced in the 1 g environment on the Earth. Under technology experiments the assembly of large space structures will be possible. In addition, the transfer of fluids (cryogens, propellants, etc.) will be investigated since this becomes a nontrivial operation in orbit. While satellite retrieval and minimal servicing is possible with the Shuttle, space operations such as satellite servicing and platform servicing will become commonplace.

IV. Experimental Utilization

The next figure (figure 4) addresses the utilization aspects of the Space Station. The Space Station is to be user friendly, i.e., the complications of flying an experiment will be reduced significantly compared to the current situation. From the beginning, the Space Station Program has been focused upon the user, and utilization continues to be an important element of the program. As has been discussed previously, the users span science, commerce, and technology. User requirements are helping to shape the system requirements of the Space Station although they are not driving the system requirements due to constraints such as cost. It is expected that the Space Station design will accommodate a performance envelope (power, weight, volume, etc.) which will be derived from time-phased mission models from each user category.

Obviously, the environment to which an experiment would be subjected is very significant. Some of the characteristics[3] of this environment are indicated in figure 5. The planned altitude of the Space Station orbit is 500 kilometers (≈270 nautical miles); the planned inclination of the orbit is 28.5°. The pressure at this altitude is shown for both sunspot minimum and maximum. These pressures would be important for externally-attached payloads. The internal pressure in the laboratory modules is under investigation and will be decided at a later time. The planned protection probability from meteroids or debris is 95% within the Space Station modules. External to the Space Station ultraviolet and particle radiation will be encountered as shown in the figure. In the central portion of the figure are shown the desired gravitation levels. At present a level of 10^{-6}g is the desired level in the vicinity of the center of gravity.

Up to the present time the Space Station Program has supported the conceptual definition of technology experiments. The description and details of each experiment resides in the Space Station Mission Requirements Data Base.[4] Technology development mission (TDM) is the term used for the conceptual technology experiments. There are approximately 70 TDM's in the Mission Requirements Data Base. These TDM's for the most part consist of NASA experiments. The TDM's have been categorized into six areas[5] as shown in figure 6. These categories are materials and structures, energy conversion, communications and electronics, propulsion, controls and human factors, and systems operations. The materials and structures category contains materials performance and processing, deployment/assembly, construction, and structural dynamics. The energy conversion category encompasses solar concentrators, laser power transmission/reception, waste heat rejection, and power subsystems. The TDM's under the communications and electronics category include space antennas, telecommunication systems, space interferometer systems, and Earth observations. The propulsion category covers fluid management and low thrust propulsion. Under the controls and human factors category are figure controls and devices, information systems, teleoperation, and interactive human factors. The systems operations category contains environmental effects, habitation, medical, tether systems, satellite and OTV servicing, and systems operations.

V. Technology Development Mission Examples

The following examples are technology development missions which are representative of each category in the technology portion of the Mission Data Base.

TDM's numbering 2000-2099 fall into the materials and structures category. Representative of this category is TDM 2071 entitled Flight Dynamics Identification.[6] Figure 7 shows a conceptual drawing of the experiment. The following is a brief statement of the objective and description of this experiment:

TDM 2071 - Flight Dynamics Identification

OBJECTIVE:

The purpose of this flight dynamics experiment is to develop the technology necessary to perform autonomous, in-space system identification, including the capability to estimate the shape, orientation, surface quality, flight dynamics parameters, and mass properties of various components of the Space Station (e.g., beam structures, deployable antenna assemblies and solar panels).

DESCRIPTION:

The conceptual definition of the Flight Dynamics Identification Experiment has been developed for implementation on the Space Station. Mission experiment will center upon an antenna reflector attached to a boom structure deployed on the Space Station. Retroreflector targets, typically mirrors, will be placed on the antenna surface. A laser beam will illuminate the antenna surface, and the retroreflectors, and a sensor will detect deformations in the antenna surface from reflected light.

The energy conversion category contains TDM's numbering 2100-2199. Representative of this category is TDM 2153 entitled Solar Dynamic Power Test Facility,[7] a conceptual rendering of which is shown in figure 8. This technology development mission has the following objective and description.

TDM 2153 - Solar Dynamic Power Test Facility

OBJECTIVE:

To provide a dedicated area on Space Station for flight evaluations and test operation of candidate solar dynamic power systems, subsystems and components. The flight evaluation work would be separate and apart from the operational power systems providing power to the station.

DESCRIPTION:

Solar dynamic power systems consist of solar collectors, heat receivers, dynamic power conversion systems, and radiators.

Several candidates for each component could be tested either individually or as part of a complete system. Solar collectors could be tested with different reflective surfaces (aluminum, silver) with different optical configurations (simple, parabola, cassegrainian). Solar receivers could have different heat storage materials, different operating temperatures and could be tested including Rankine, Brayton, and Stirling thermodynamic cycles. Space radiators tested could include tube and fin radiators, heat pipe radiators, and advanced radiator concepts. A major objective of the facility would be to perform configuration tests of a variety of advanced components and configurations that have shown promise in ground testing of improved Station power.

Technology development missions in the communications and electronics category are numbered 2200-2299 in the Mission Data Base. TDM 2441 entitled Microelectronics Data System[8] is representative of this category regardless of its current number in the Mission Data Base. A sketch of this experiment is illustrated in figure 9. This TDM has the following objective and description.

TDM 2441 - Microelectronics Data System

OBJECTIVE:

To operate in a realistic space environment the microelectronic, optical and opto-electronic components of advanced, high-data-rate data systems in order to establish the space worthiness of the technology; including data bus technology and data transmission in the microwave bandwidth region.

DESCRIPTION:

The experiment will develop a long-term data base on the performance of advanced microelectronic and opto-electronic data system technology for the Space Station and other space systems. It will have the flexibility to incorporate new and developing technologies, including gallium arsenide switching, integrated optic memory, and data processing. The experiment will test concurrently three independent data system/component modules. The experiment will operate semi-autonomously and be externally deployed. The mission will consist of a 5- to 10-year program of experiments, with annual recovery of exposed/tested modules and installation of new modules. The Space Station will provide the means for necessary long-term exposure to the space environment (including low-dose-rate effects).

The propulsion category contains TDM's numbering 2300-2399 in the Mission Data Base. TDM 2311 entitled Long-Term Cryogenic Fluid Storage[9] is characteristic of the experiments in this category. A diagram of this TDM is shown in figure 10. The following objective and description are given to convey some understanding of this experiment.

TDM 2311 - Long-Term Cryogenic Fluid Storage

OBJECTIVE:

To develop insulation and refrigeration system technology to provide long term orbital storage of cryogenic liquids.

DESCRIPTION:

Subscale cryogenic fluid storage tanks and refrigeration systems would be tested to establish thermal performance and useful life during the early phases of the Space Station evolutionary process. Selected concepts will then provide design criteria for cryogenic fluid storage and supply systems to provide Space Station consumables and orbit transfer vehicle propellants.

TDM's in the controls and human factors category are numbered 2400-2499 in the Mission Data Base. TDM 2411 entitled Advanced Adaptive Control[10] is representative of the experiments in this particular category. A conceptual rendering of this experiment is shown in figure 11. This TDM has the following objective and description.

TDM 2411 - Advanced Adaptive Control

OBJECTIVE:

The underlying objectives are to develop, demonstrate, and evaluate flight system performance and stability improvement; sensing strategies and mechanization; control gain update subroutines and reconfiguration schemes; and adaptive control algorithms. Included in this mission are the development of applied adaptive control concepts that will be implemented and mechanized as algorithms for compensation of gross system model uncertainties and changes and the demonstration of autonomous error estimation and adaptive control techniques that are needed for compensation of inevitable system and model uncertainties during space payload deployment.

The specific objectives include assessment and verification of design and performance effectiveness using the adaptive control algorithms for systems control in the presence of parameter uncertainties and variations onboard the Space Station. This document addresses the design of the mission experiments from both the equipment/instrumentation and functional aspects.

DESCRIPTION:

A mission of 90 operational days is planned. The mission equipment includes computers/data processors and sensor monitors that are common to other TDMX's. All experiments for this mission will involve the use of an antenna structure that is comprised of a reflector supported by a boom geometry consisting of a long and short boom. A six-degree-of-freedom (DOF) gimballed torquing device, on which the long boom will be mounted and firmly attached, will provide rotations and mechanical excitations for the antenna structure. A set of forcing devices may be attached to different parts of the two booms to enhance and increase vibrations through excitations; sensors and actuators will be placed on the reflector and the booms. The sensors will detect forcing disturbances and figure or shape distortions. The actuators will, upon control action through the adaptive control process, suppress those vibrations through damping which will restore structural figure and shape. The adaptive control algorithms will be present in control software packages in the mission computers employed for this experiment. Because algorithm size and complexity may be considerable, usage of the Station computer may, for certain experiments, be necessary. A mission specialist will be responsible for monitoring and conducting the experiments. Experimental data generated from this experiment is then transmitted to ground stations for analysis. The data base acquired, and subsequent analyses, will be used to assess and evaluate the effectiveness and responsiveness of adaptive control techniques used for antenna figure and pointing control, and to analyze sensor and actuator performance in terms of generic controllability.

Finally, in the systems operations category, the TDM's are numbered 2500-2599 in the Space Station Mission Data Base. TDM 2572 entitled Cryogenic Propellant Transfer, Storage, and Reliquefaction Technology[11] is representative of the set of experiments in this category. A diagram of this experiment is shown in figure 12. This technology development mission has the following objective and description.

TDM 2572 - Cryogenic Propellant Transfer, Storage and Reliquefaction Technology

OBJECTIVE:

To test and verify the hardware and techniques developed to reliquify cryogenic propellant boil-off and to establish an accurate data base for accomplishing propellant transfer, storage, and reliquefaction for long periods of time in space.

DESCRIPTION:

The system consists of supply, receiver, and refrigeration components. Propellant transfer is done by using a pump with a full screen propellant acquisition device. The supply tank contains subcritical fluid and requires an acquisition device for providing liquid to the transfer line. During reliquefaction the following will be accomplished: (1) perform parametric thermal testing to determine performance of passive storage and active refrigeration equipment; (2) determine refrigeration or reliquefaction capacity, power requirements, heat rejection, efficiency, boil-off, stability, automatic control; (3) perform tests on compressors, expansion process, heat exchangers, etc.; and (4) determine fluid leakage, particle freezeout, contamination.

VI. In-Space Research, Technology, and Engineering

Currently, the Office of Aeronautics and Space Technology (OAST) at NASA Headquarters is undertaking an in-space research, technology, and engineering (RT&E) program to establish candidate activities for 1990 and beyond. This program encompasses the technology experiments which are precursors to Space Station experiments and technology experiments to be performed on the Space Station itself. The initial phase of this in-space RT&E program will drive out the actual user requirements since it addresses industry, university, and other Government users in addition to the technology users identified within NASA itself. OAST is approaching the technology users of the Space Station by means of a technology experiment theme approach.

Initially, five themes have been identified for technology experiments. They are listed in figure 13. The themes include space structure (dynamics and control), energy systems and thermal management, space environmental effects, fluid management, and in-space operations. The OAST is conducting the In-Space Research, Technology, and Engineering Workshop at the National Conference Center at Williamsburg, Virginia, on October 8-10, 1985, where the themes will be validated, changed, or expanded as a result of recommedations from the technology user community. Presently, the space structure (dynamics and control) theme includes advanced structural concepts, structural dynamics, advanced control concepts, structure/control interaction, and structure/control sensors. The energy systems and thermal management theme covers advanced photovoltaics, solar dynamics, nuclear, advanced thermal concepts, and laser power. Under the space environmental effects theme fall material durability, plasma, and contamination. The fluid management theme includes fuel storage and transfer, fluid behavior, and sensor concepts. The in-space operations theme contains automation and robotics, sensor techniques, information systems, advanced life support systems, tethers, orbital transfer vehicle, system testing, and propulsion.

VII. Concluding Remarks

The activities centered around the technology users of the Space Station are extremely important and timely since the accommodations required for technology experiments need to be identified early in the definition and design phase of the Space Station Program. Also, the driver missions need to be identified, i.e., those technology experiments which require significant power, volume, data rates, etc. From the identification of credible technology experiments, it will be possible to generate an envelope of technology experimental requirements as a function of time. This effort will support not only the planning for the initial Space Station but also the growth version of the Space Station.

VIII. References

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IX. Acknowledgements

The authors gratefully acknowledge Dr. Susan M. Benford of the NASA Lewis Research Center, John C. Mankins of the Jet Propulsion Laboratory, and John W. Maloney of General Dynamics Corporation for providing graphics related to the technology development missions. The authors also wish to thank Jane A. Hagaman for her valuable assistance. Finally, the authors express their appreciation to P. Kay Millen for her tireless efforts in typing the manuscript and reproducing many of the figures.

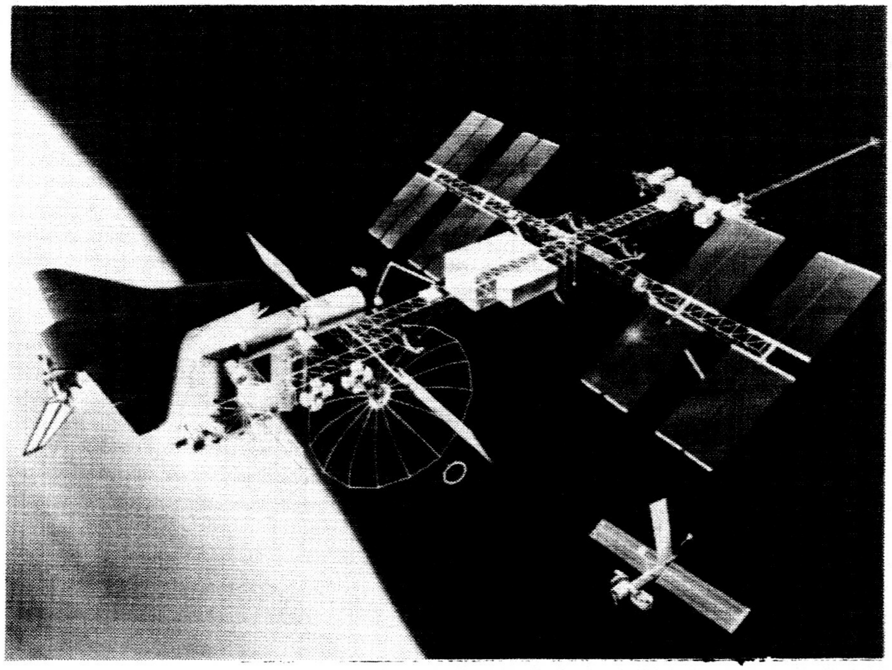

Figure 1. Reference Configuration of Space Station

- A NATIONAL LABORATORY IN SPACE
- A PERMANENT OBSERVATORY
- A SERVICING FACILITY
- A TRANSPORTATION NODE
- AN ASSEMBLY FACILITY
- A MANUFACTURING FACILITY
- A STORAGE DEPOT
- A STAGING BASE

Figure 2. Purposes of the Space Station

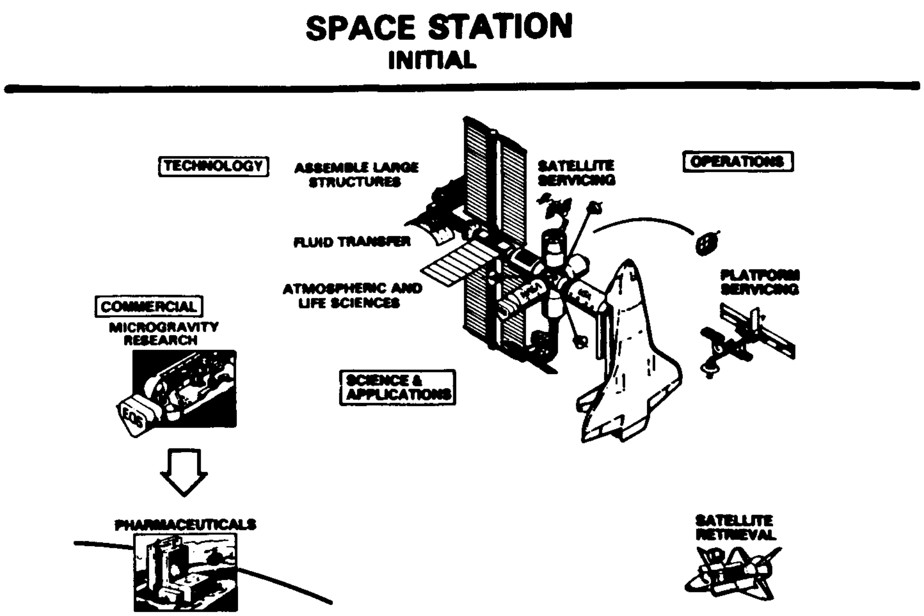

Figure 3. Uses of Initial Space Station

- SPACE STATION IS TO BE USER FRIENDLY
- FROM THE START, PROGRAM FOCUSED UPON UTILIZATION
- USERS: SCIENCE, COMMERCE, TECHNOLOGY
- USER REQUIREMENTS HELP TO SHAPE SYSTEM REQUIREMENTS
- PERFORMANCE ENVELOPE

Figure 4. Utilization Aspects of Space Station

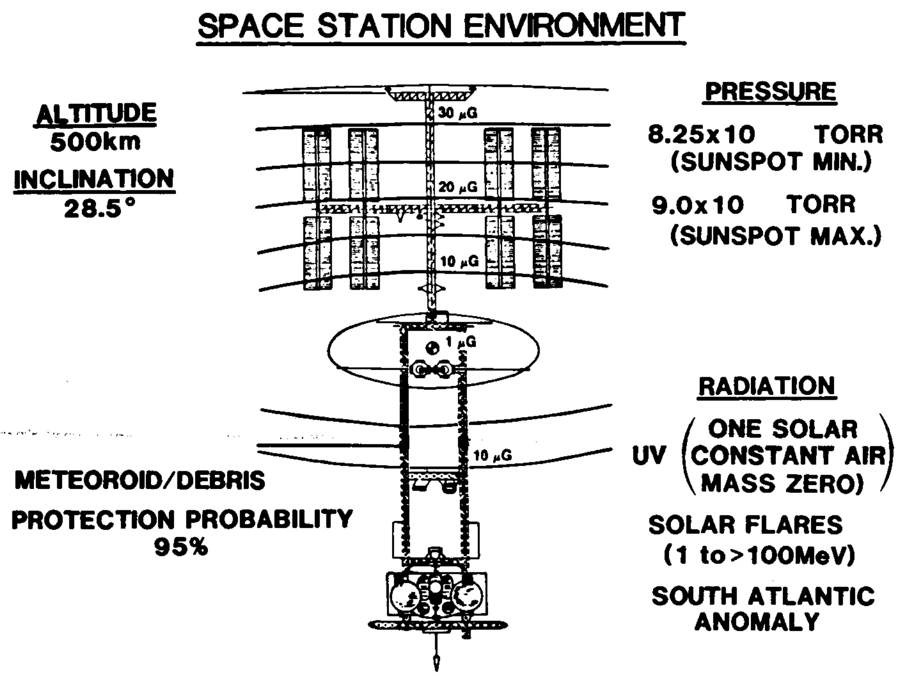

Figure 5. Characteristics of Space Station Environment

- **MATERIALS AND STRUCTURES**
- **ENERGY CONVERSION**
- **COMMUNICATIONS AND ELECTRONICS**
- **PROPULSION**
- **CONTROLS AND HUMAN FACTORS**
- **SYSTEMS OPERATIONS**

Figure 6. Categories of Technology Development Missions

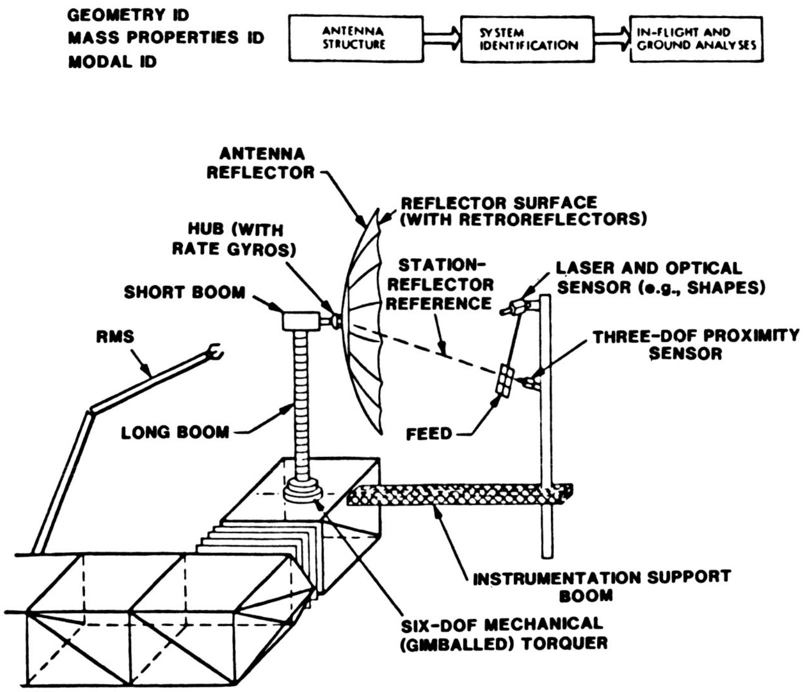

Figure 7. TDM 2071 - Flight Dynamics Identification

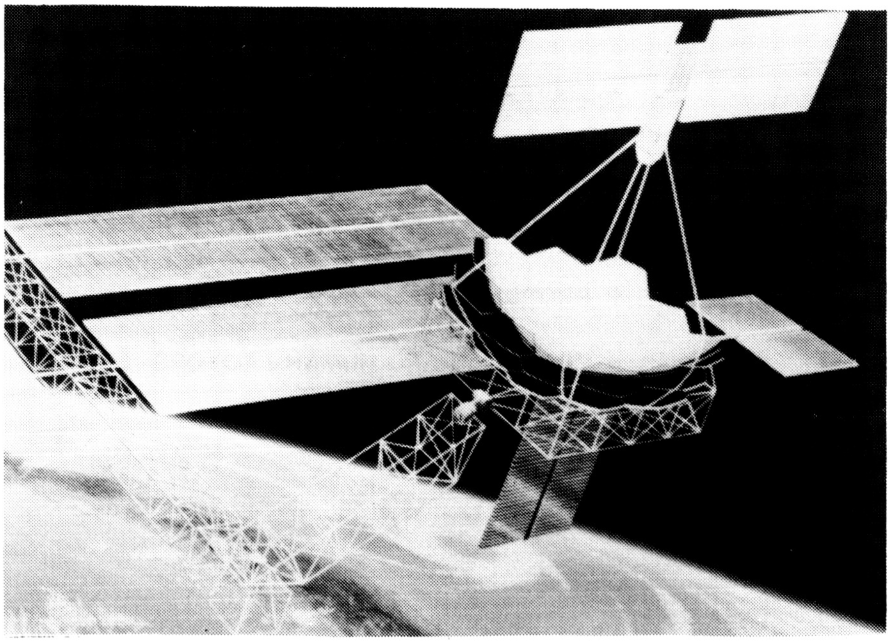

Figure 8. TDM 2153 - Solar Dynamic Power Test Facility

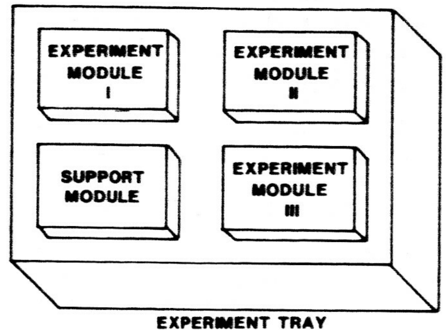

Figure 9. TDM 2441 - Microelectronics Data System

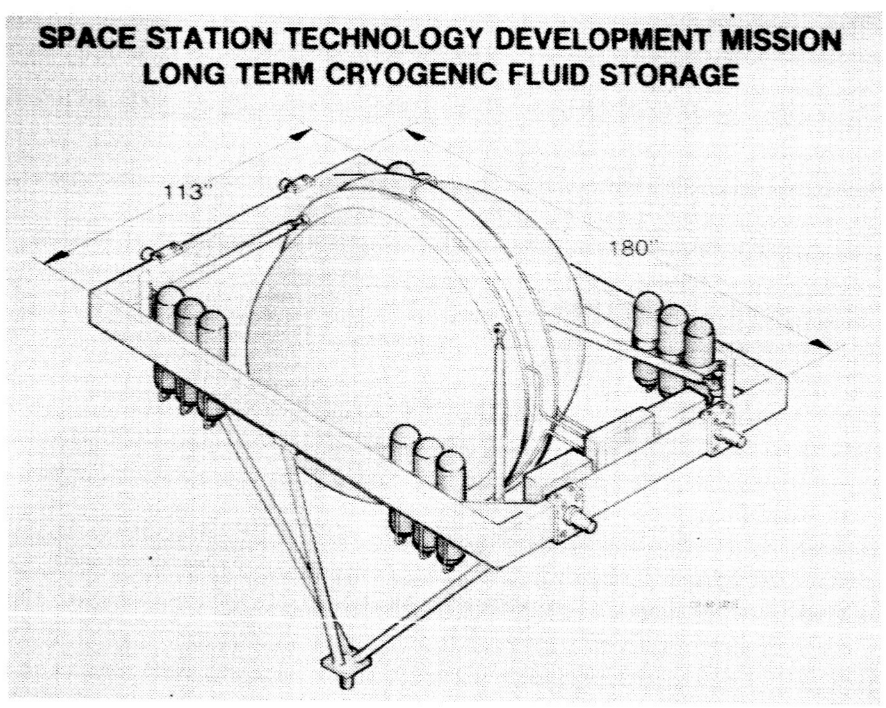

Figure 10. TDM 2311 - Long-Term Cryogenic Fluid Storage

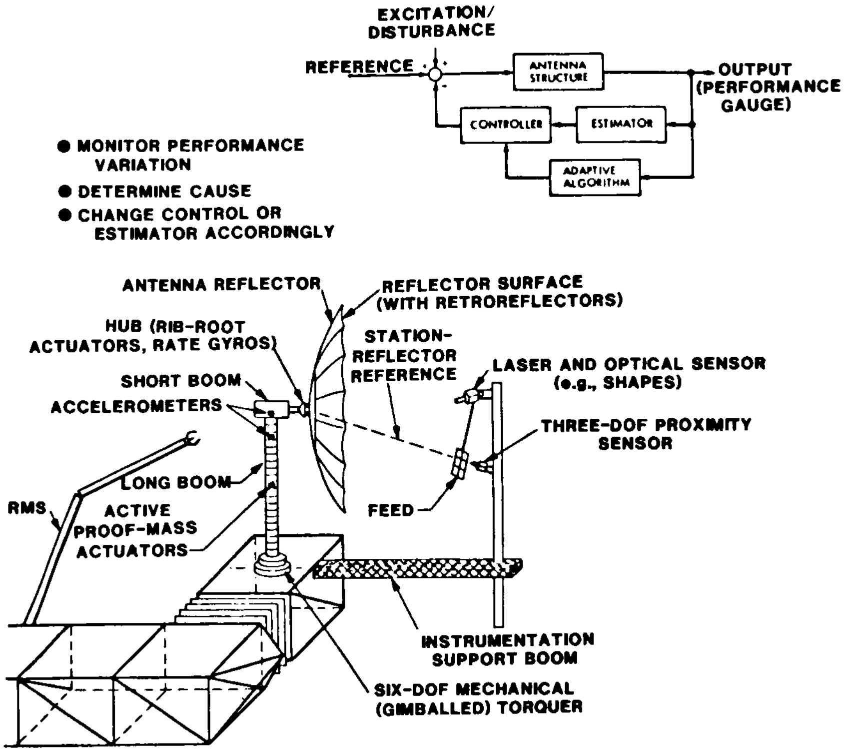

Figure 11. TDM 2411 - Advanced Adaptive Control

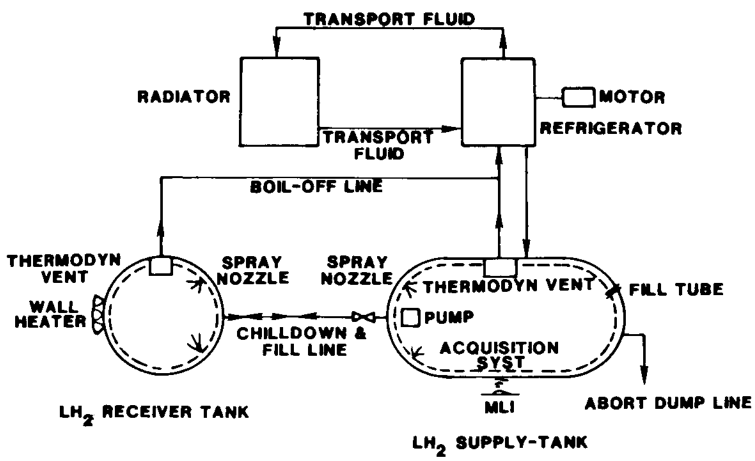

Figure 12. TDM 2572 - Cryogenic Propellant Transfer, Storage and Reliquefaction Technology

- SPACE STRUCTURE (DYNAMICS AND CONTROL)
- ENERGY SYSTEMS AND THERMAL MANAGEMENT
- SPACE ENVIRONMENTAL EFFECTS
- FLUID MANAGEMENT
- IN-SPACE OPERATIONS

Figure 13. In-Space Research, Technology, and Engineering Themes

SPACECRAFT DESIGN FOR SERVICING*

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Abstract

A large number of institutions addressed the design of spacecraft for on-orbit servicing when the Space Shuttle Program was being started. The resulting extensive literature resource was used to arrive at a preliminary design of an on-orbit servicer and compatible design concepts of representative serviceable spacecraft. This discussion describes the design concepts and presents some general conclusions and recommended approaches. It is not difficult to design spacecraft for serviceability once the spacecraft project and the designers decide to do so. The associated weight and cost penalties were estimated to be small (cost increments of 4% for design and development and 8% for unit cost). Two additional areas for technology application are also discussed.

Introduction

One of the justifications for the Space Transportation System was its potential for supporting the repair or recovery of failed spacecraft. This approach was extended to the concept of making less expensive spacecraft, accepting the higher predicted failure rates, and using the Shuttle to permit repair of those spacecraft that did fail. This spawned a large number of government, academic, and industry studies on how spacecraft might be configured for on-orbit servicing. Figure 1 illustrates the variety of concepts that were documented. The whole gamut from recovery and ground refurbishment, through repair at the Orbiter, through remote operations in low earth orbit, to repair in geosynchronous orbit were addressed. All of the concepts we discuss these days were addressed then except for Space Station related operations. The long cylindrical spacecraft represents the Space Tug whose missions are now to be handled by the Orbital Maneuvering Vehicle and the Orbital Transfer Vehicle.

* For presentation at Satellite Services Workshop II

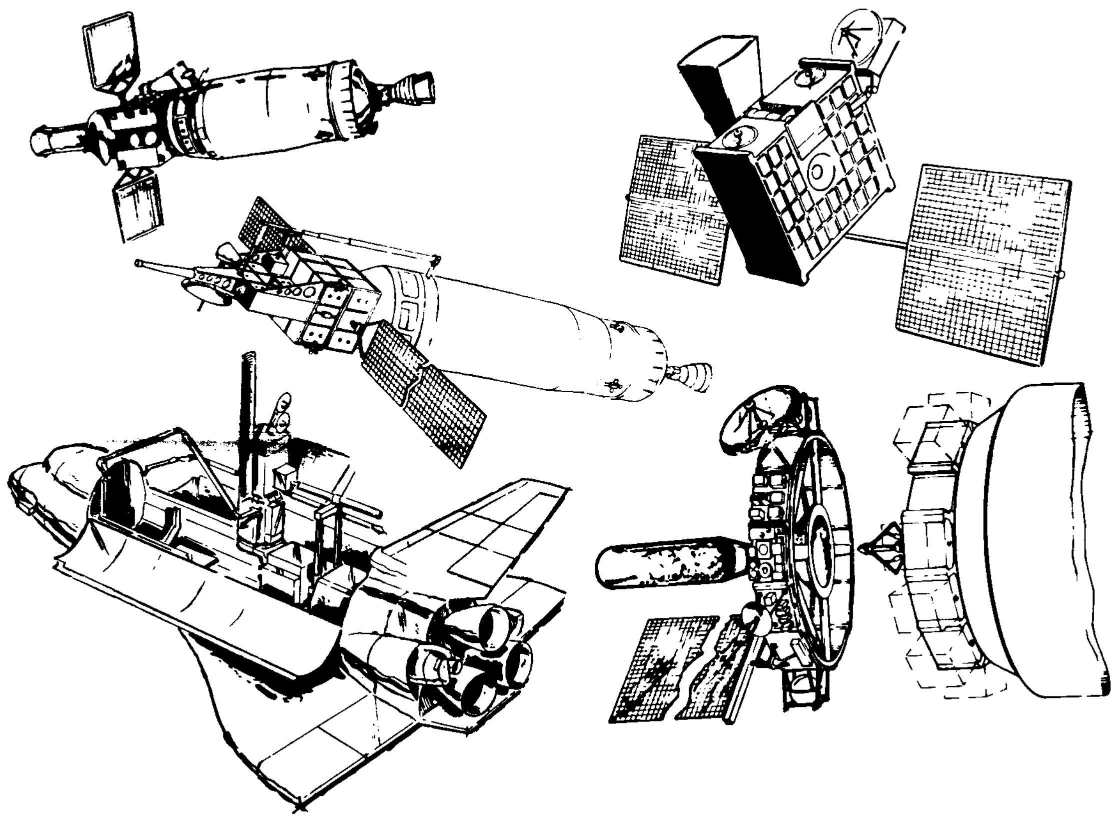


Figure 1 Serviceable Spacecraft Designs From the 70's

The extensive resource base was used in a 1974 through 1978 study conducted by Martin Marietta for Marshall Space Flight Center. Some of the results of that work were included in the presentations at this workshop by Don Scott and Jim Turner of MSFC. A good summary of the early work is given in Proceedings of the Second Conference on Payload Interfaces, MDC G4818, McDonnell Douglas Astronautics Company, Huntington Beach, California, September 6-7, 1973.

Table 1 lists those factors that were used to form the basis for design of a spacecraft servicing system. These factors were not selected a priori but evolved as the study progressed or were a consensus from the literature. Module (or On-orbit Replaceable Unit) exchange was selected as the major servicing activity. A module is thought of in a more general sense than just an electronics package. It can be a piece of experiment equipment, a set of thrusters, a tank of propellant, a communications antenna, or even a fluid umbilical connection. This broad interpretation of "module exchange" increases the percentage of spacecraft faults that can be repaired by this technique.

Table 1 Spacecraft Servicing Design Basis

- MODULE EXCHANGE IS A MAJOR SERVICING ACTIVITY
  - FAILED ORU REPLACEMENT
  - EQUIPMENT UPGRADE
  - FLUID RESUPPLY
  - PRODUCT RETURN
- ALLOCATE REQUIREMENTS BETWEEN SERVICER AND SPACECRAFT
- SPACECRAFT PROGRAM REQUIREMENTS
  - ON-ORBIT REPLACEABLE UNITS (ORU)
  - ELECTRICAL UMBILICAL FOR STATUS AND CONTROL
  - FAILURE ISOLATION TO AN ORU (WITH GROUND SUPPORT)
- SERVICER SYSTEM
  - TRANSPORTS ORUs
  - EXCHANGES ORUs
  - RESUPPLIES FLUIDS
  - HANDLES ADAPTERS AND TOOLS
- TRANSPORT TO ORBIT BY SHUTTLE
- LAUNCH COST MAGNITUDE IMPLIES MISSION PREPLANNING

The second item in Table 1 implies the willingness to allocate functional requirements to the spacecraft as well as to the servicer system. The few functions assigned to the spacecraft are very important because of how they can simplify the design of the servicer system. This type of spacecraft could be called servicing compatible. The last line of the table is also significant in that it implies the mission planner should have high confidence that his planned servicing mission will succeed or else he may be wasting tens of millions of dollars. Preplanning means having faults isolated to an ORU and taking a good replacement ORU along on the servicing mission. It also implies that the entire geometry of the ORU replacement can be preprogrammed. Only uncertainties in geometry due to the docking system, the servicer system, and thermal effects need to be accommodated. Also any required special tools or adapters can be taken along.

The spacecraft servicing design approach items listed in Table 2 evolved during the course of the study. The selected servicer system can be applied to most spacecraft that will be launched in the Shuttle because its size, degrees of freedom, and joint ordering were carefully selected to match this class of spacecraft. In some cases more than one docking may be necessary and good judgement should be used in locating the modules and their attachment interfaces.

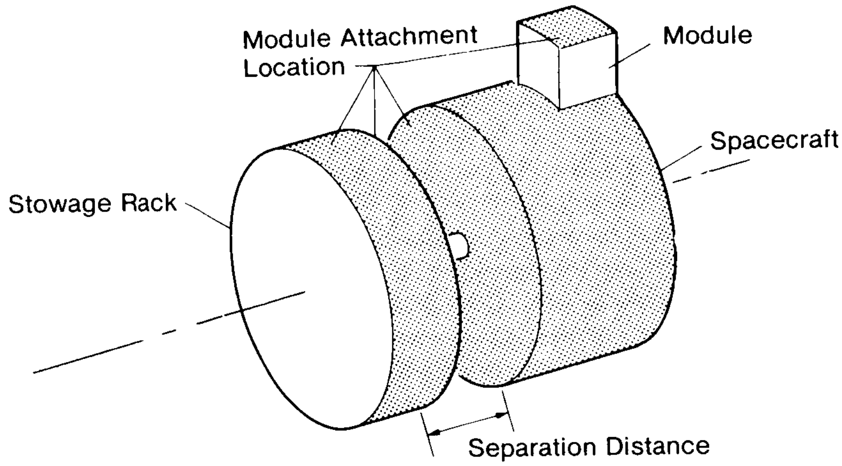
Table 2 Spacecraft Servicing Design Approach

• APPLICABLE TO MANY SPACECRAFT • INCORPORATE EXISTING TECHNOLOGY • PLACE FEW REQUIREMENTS ON SPACECRAFT • REMOTELY OPERABLE - AUTONOMOUS FOR MOST ACTIVITIES - TELEOPERATION AS ALTERNATIVE	• ORU INTERFACES OPERABLE BY - SERVICER SYSTEM - ASTRONAUTS ON EVA - GROUND HANDLING EQUIPMENT • SUPPORTED BY CARRIER VEHICLE - RENDEZVOUS AND DOCKING - ATTITUDE CONTROL - ELECTRICAL POWER - TWO-WAY COMMUNICATIONS

The decision to go with existing technology was easy to make because advanced technology is not required. Some forms of Artificial Intelligence could be useful for the planning operations. In particular, an expert system could be used to help isolate faults to specifics ORUs and a planning system could be used to interface with CAD/CAM representations of the failed spacecraft and the servicer system to develop the data required for automatic module exchange trajectory generation. As the servicer system must be transported to the failed spacecraft by a carrier vehicle - Orbiter, Orbital Maneuvering Vehicle, or Orbital Transfer Vehicle - and each of these carrier vehicles can supply certain support functions, it was decided to rely on the carrier vehicle to provide the support functions listed at the bottom of the table. The Space Station can provide the last three support functions, so it could be used as a carrier vehicle for on-orbit servicing if the failed spacecraft, or equipment, could be brought to the servicer.

Servicer System

It is useful to discuss the servicer system before the serviceable spacecraft, as its form evolved first. A wide variety of servicer mechanism configurations were identified in the literature. They ranged from simple one degree-of-freedom (DOF) devices, through a three DOF rectangular travel system, to two-arm concepts, each with 7 DOF. The selected approach started with the Shuttle launch cost rules that favored flat disk-shaped spacecraft such as the Orbital Maneuvering Vehicle (OMV). From this, the servicer working volume and observations shown in Figure 2 were developed.



Observations:

- The module attachment locations form a surface of revolution about the spacecraft centerline.
- The first servicer degree of freedom should be roll about the base of the docking probe.
- The need for minimum arm length and separation distance implies the servicer mechanism must "reach around" the spacecraft and module surfaces.

Figure 2 - Servicer Mechanism Working Volume

The shaded area on Figure 2 represents the regions where the servicer mechanism end effector must reach. The direction of module removal is generally perpendicular to the shaded surface. The applicability of a roll rotation for the first degree of freedom is quite apparent. As the separation distance between the spacecraft and stowage rack is reduced, the space available for servicer mechanism elements near the base is reduced and the "reach-around" problem becomes more difficult. The minimum separation distance was taken as 60 in. which allows for a 40-in. module, a ten-in. end effector, and a five-in. clearance on each end. The "reach-around" problem leads to use of a redundant degree of freedom.

Figure 2 implies that two layers, or tiers, of modules could be incorporated at a single docking location. It was later decided to simplify the servicer design to permit module exchange only from the first tier and to wait until a specific need is identified before the servicer configuration is grown to handle the second tier.

An extensive review and analysis of servicer mechanism configurations and 28 serviceable spacecraft configurations was performed to arrive at the selected servicer configuration shown in Figure 3. From the review and analysis, extensive sets of requirements were prepared and refined. All servicer configurations involving one or two arm segments and many three arm segment configurations were considered.

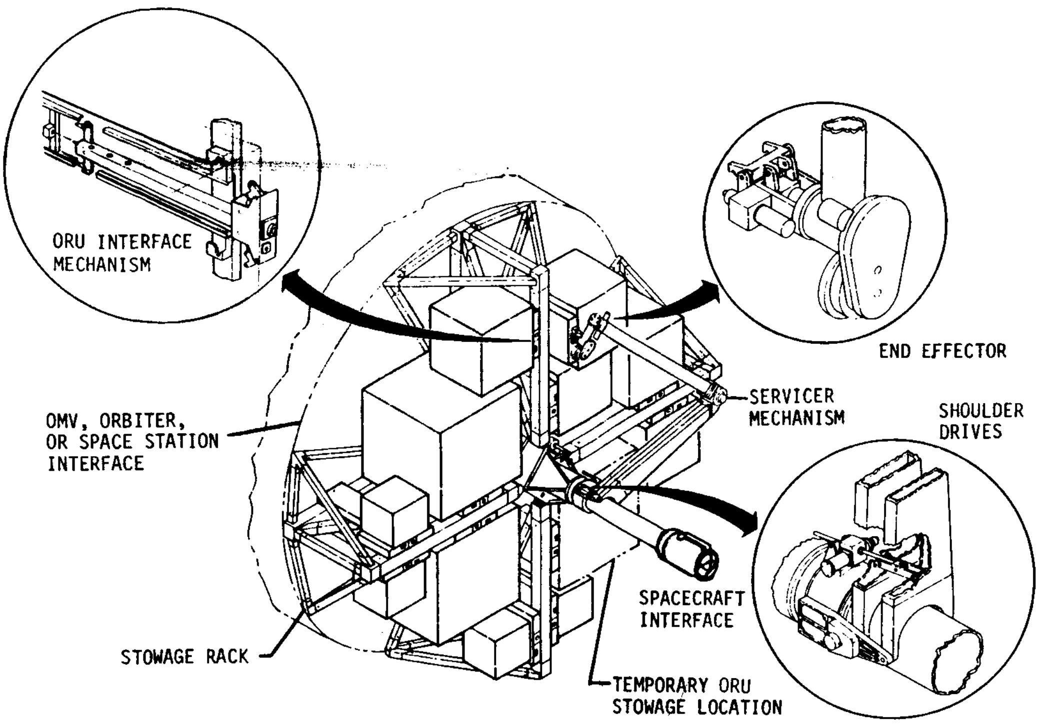


Figure 3 Integrated Orbital Servicing System (IOSS)

This design has only two major components: (1) a servicer mechanism, and (2) a stowage rack for module transport. A docking mechanism is also shown for reference. The servicer mechanism and the stowage rack were designed separately with interfaces for individual removal and replacement. Stowage racks can be configured and loaded for particular flights prior to attachment to the carrier vehicle. It may be desirable to have available several stowage racks for this purpose. The stowage rack shown mounts directly to an upper stage such as the Orbital Maneuvering Vehicle. A flight support structure has been designed to adapt the stowage rack shown to the Orbiter.

The entire design of the servicer system has been predicated on the simple nature of the module exchange task as compared to the broader variety of tasks that a general purpose manipulator is called upon to perform. The simple activities of remove, flip, relocate, and insert modules, when combined with the facts that all aspects of the module trajectories are known far in advance of use and that the work volume is a simple solid of revolution, have been used in many ways to result in a basically simple design in terms of mechanism configuration, control system design and operations approach. This simplicity was accentuated by performing the mechanism and control system designs concurrently in an integrated manner so that each of the needed functions was allocated to the system that could most effectively accomplish it.

Three modes of control were included. The Supervisory mode of control was proposed as the normal mode of operation. All servicer arm motions and trajectories are determined before flight and stored on board. A Manual-Direct mode is provided as a totally unsophisticated means of backup control. It sends commands directly to the joints themselves. The Manual-Augmented mode has man doing most of the arm control as in the Manual-Direct mode only using hand controllers instead of panel switches.

The physical attachment between an ORU and the spacecraft or stowage rack is called an interface mechanism. A representative side interface mechanism is shown in Figure 4 with and without a module representation. The mechanism uses a three point, nonredundant, attachment system so spacecraft thermal and structural loads do not pass through the module. The bell crank linkage is driven via a worm and gear from a motor on the end effector. A spring-loaded self-aligning tongue in a slot accomplishes the mechanical interface. The linkage starts engagement with a low force that gradually increases to 200 lb as the links approach an over-center position. Total travel is 1-3/4 inches.

The study suggested the development of an interface mechanism as a two-part kit in perhaps three sizes. These standard interface mechanisms could be made available to spacecraft designers. Each designer could then make his choice within his own set of design and economic constraints. The graph on the facing page is a histogram from data on 683 modules from 30 different

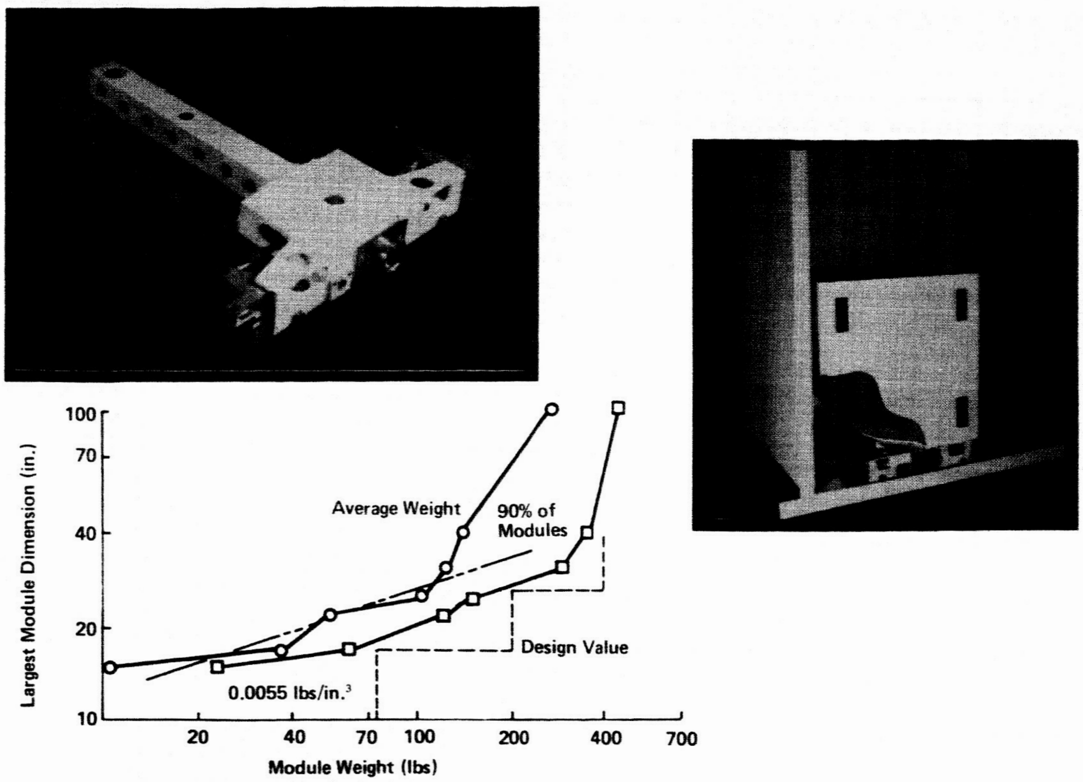
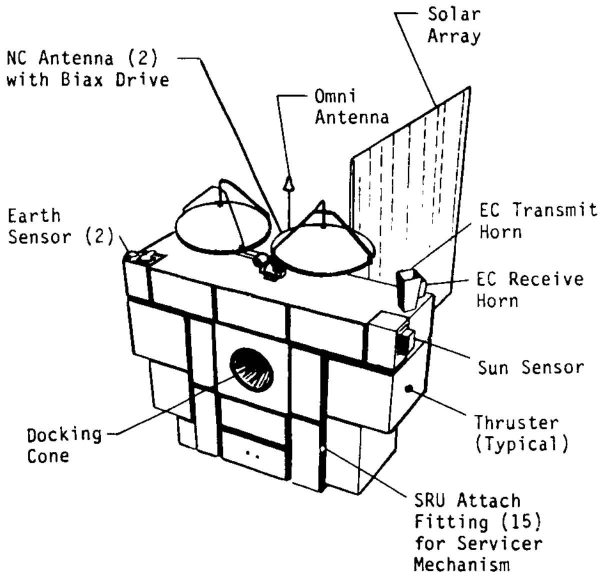


Figure 4 Side Mounting Interface Mechanism

serviceable spacecraft. The recommended interface mechanism standard sizes thus became-- 17 in., 26 in., and 40 in. These correspond to modules no larger than a cube of the indicated dimension. The recommended corresponding module weight limits are shown on the graph.

Serviceable Spacecraft Designs

The serviceable communications satellite shown in Figure 5 is one of three serviceable spacecraft designs prepared by TRW, Inc. It is an excellent representative of the form all geosynchronous communications satellites might take. It is a single tier and is box shaped. All modules are removed axially. In this configuration, a single solar array mounted opposite the docking port is used. An advantage of the configuration shown is that the exposed faces of the modules, or on-orbit replaceable units (ORUs), see very little of the sun and thus can be used to radiate heat out of the modules.



- GEOSYNCHRONOUS ORBIT
- BOX SHAPE
- 99 IN. X 128 IN. X 40 IN.
- WEB STRUCTURE
- SINGLE CENTRAL DOCKING
- SINGLE TIER
- AXIAL MODULE REMOVAL
- 15 MODULES
- LARGEST -- 40 IN. X 40 IN. X 32 IN.
- HEAVIEST -- 444 LBS
- MINIMUM REACH -- 23 IN.
- MAXIMUM REACH -- 72 IN.

Figure 5 Serviceable Communications Satellite

A breakdown of the spacecraft mass properties shows that 1,920 lb, or 81% of the total spacecraft weight, is space-replaceable. The major items that are not serviced are the basic structure, the solar array (the solar array drive is replaceable), the narrow-coverage antennas and their biax drives, the horn antennas, the omni antenna, and the shunt element assembly. The spacecraft structure is designed to maximize the volume available for components to be carried in the ORUs, to maximize radiator area for thermal control, and to interface with the servicer. This type of structure is less efficient than those designed for expendable spacecraft, but not by a great amount. The docking cone for servicing is located in the center bay of the egg-crate-like structure. The walls of this bay form a fully-closed box, as do all the internal ORU mounting structures. The walls are one-inch thick honeycomb core sandwich panels. Tubular support struts are located on each side to help support the wide upper structure.

The second of the TRW serviceable spacecraft designs was the Synchronous Earth Observatory Satellite (SEOS). The SEOS configuration (Figure 6) was defined by the large telescope involved and the location of the mission equipment. The result was also a single tier of axially removed modules. However, the docking axis was perpendicular to the telescope line of sight. The largest SEOS module involved a 60-in. dimension to provide enough area for cooling.

While greater than the usual module sizes, these modules can be carried in the spare module stowage rack.

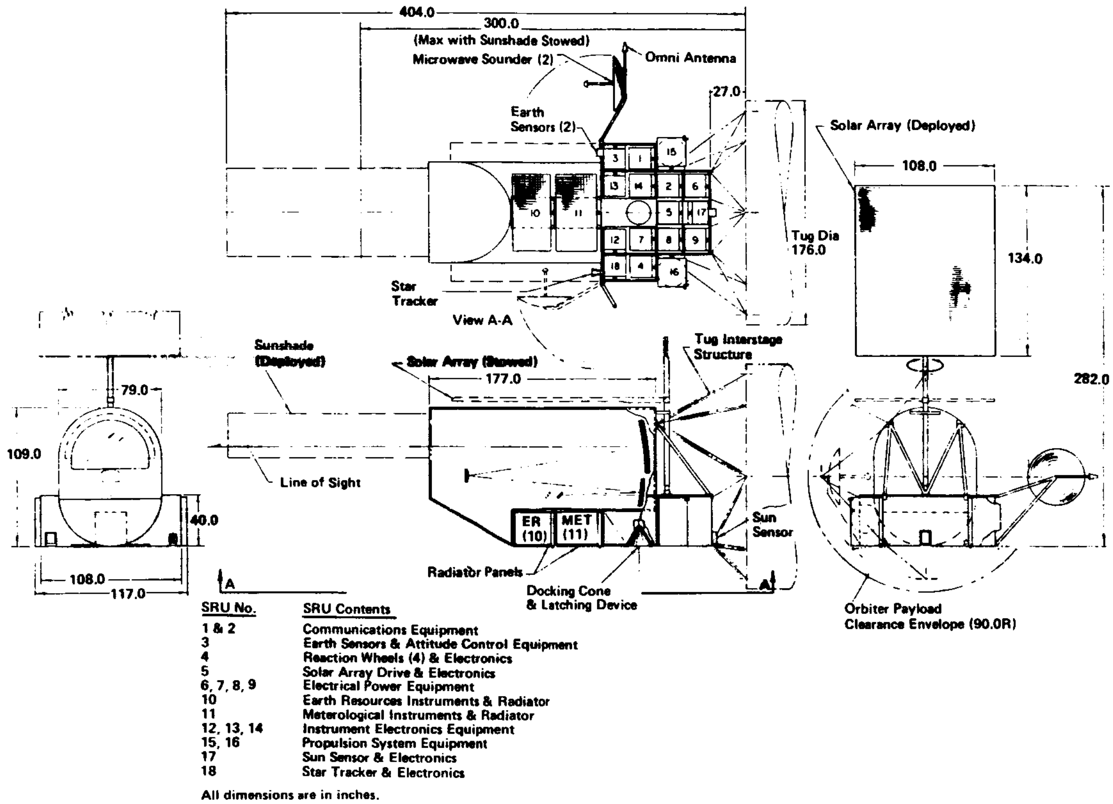


Figure 6 Serviceable Synchronous Earth Observatory Satellite (SEOS)

There had been several preliminary spacecraft design studies for the expendable and serviceable SEOS. The Large Earth Survey Telescope (LEST) system shown here was described as being capable of satisfying the earth resource and meteorological requirements. Other studies of serviceable SEOS had less complex mission equipment but they did not meet all of the performance requirements. Meteorological events to be monitored by SEOS included severe storms, hurricane and tropical storms, flash floods, frost and freeze, clear air turbulence, fog, lake and sea breezes, air pollution, and weather modification and experiment assessment.

The earth resource (ER) and meteorological (MET) instrument packages must be located on the side of the telescope for this optical system. Therefore, the docking face for all ORUs was designed to be on the same side. All of the 19 ORUs are accessible by the servicer. Side mounting interface mechanisms are used for all ORUs. The "box and shelf" type of spacecraft structure supporting the ORUs is envisioned to be of honeycomb panel construction. The solar array mast and pivot bearings are fixed to the spacecraft structure, but the drive motor and electronics are replaceable. Engagement/disengagement is provided by axial positioning of the driver/driven gear interface.

The Characteristic Large Observatory (CLO) (Figure 7) was the third serviceable spacecraft design prepared by TRW. It represents three classes of large low-earth orbit observatories -- X-ray, stellar, and solar. The stellar and solar versions were addressed in terms of their unique mission equipment. The CLO incorporates two docking ports, one aft and one forward and to the side, with the modules at each docking port arranged in a single tier. The second docking is required because the aspect sensors must be mounted at the mirror assembly and because there were too many modules to be mounted in a single tier. There are several outsize mission equipment modules, but these can be mounted in the nominal stowage rack. Roll-up thermal covers can be rolled back by the servicer end effector and then the mission equipment modules can be removed from the carousel.

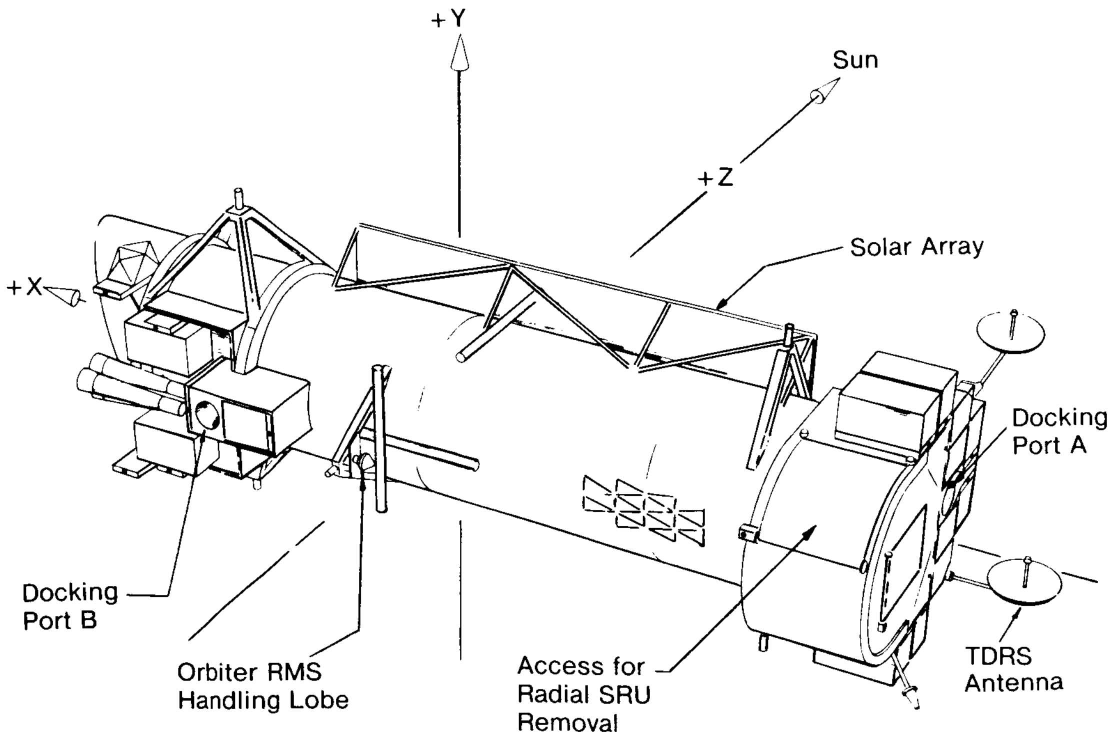
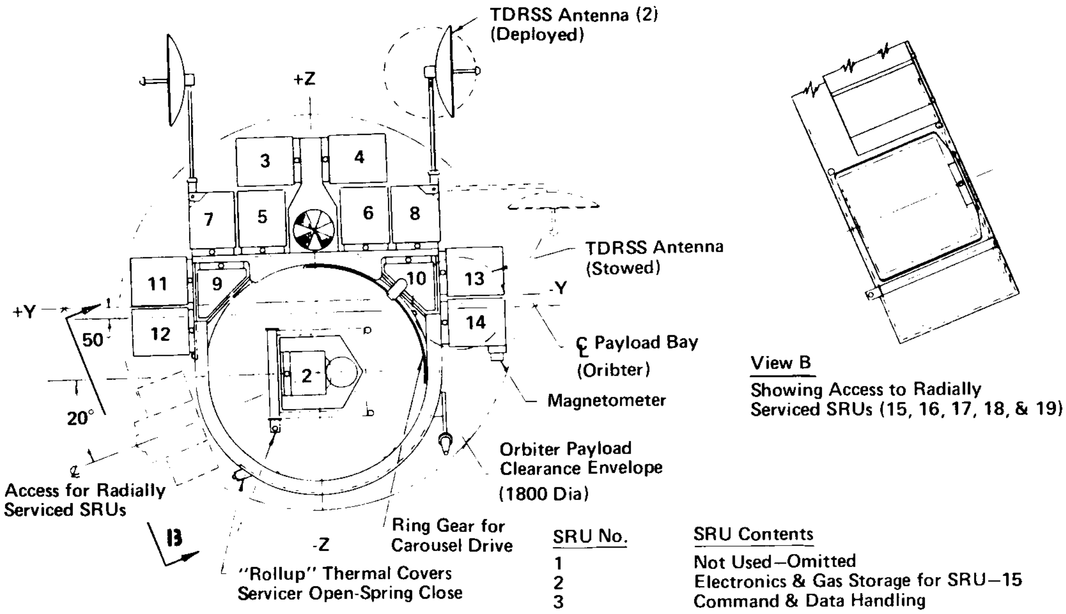


Figure 7 Serviceable Characteristic Large Observatory

The CLO represents one of the most complex spacecraft examined for serviceability. By having the servicer system designers and the spacecraft designers interact, it was possible to come up with a good serviceable CLO design in a short time. The mirror assembly and optical bench components of the telescope system are fixed in relation to the spacecraft structure. The solar array is attached to the telescope housing permanently, as are the sun sensor instruments. Five of the mission equipment instruments are mounted on a carousel which serially positions each instrument detector in the telescope focal plane. All five of these instruments are arranged in compact groups permitting each to be a replaceable unit. In order to gain access to each of the carousel-mounted ORUs, a single rotational position of the carousel is assigned, where a "door" is furnished for the servicer to perform module exchanges.

Spacecraft structure at the extreme -X end of the telescope accommodates thirteen ORU modules with uniform dimensions, plus Docking Cone A. This cone is used as the servicer docking contact for the thirteen ORUs, as well as for the five focal plane instrument ORUs on the carousel. A window shade device with thermal blanketing is used to close the opening during normal operation. The servicer end effector is used to operate a worm gear mechanism to open and close the shade. Six additional ORUs are carried at the opposite end of the spacecraft. Docking cone B is provided on the -Z side of the telescope to allow servicer access to this group. All 24 ORU use the side mounting interface mechanism.

Figure 8 is an end view of the serviceable Characteristic Large Observatory that shows most of the housekeeping ORUs. The axially-removed ORUs are numbered from 2 through 13. The ORUs on the carousel are removed radially through the access port shown in view B. The two TDRSS antennas are stowed behind other ORUs for launch of the CLO. If their ORU must be replaced, the TDRSS antenna will be deployed, at least partially. The carousel has a rim drive mechanism, located in ORU No. 10. The electronics and gas storage for the carousel-mounted focal plane crystal spectrometer are removed axially from the center of the carousel (ORU No. 2). This location also has a roll-up thermal cover actuated by the servicer end effector.



SRU No.	SRU Contents
1	Not Used–Omitted
2	Electronics & Gas Storage for SRU–15
3	Command & Data Handling
4	Spare
5, 6	Electrical Power Equipment
7, 8	TDRSS Gimbaled Antenna & Electronics
9	Digital Processor for Acds
10	Carousel Drive
11, 12, 13, 14	Reaction Wheels & Electronics

Figure 8 Serviceable Characteristic Large Observatory On-Orbit Replaceable Units at Aft Docking Port

The serviceable spacecraft designs were evaluated to obtain requirements for the servicer configuration selection. During that evaluation a number of serviceable spacecraft configuration implications became obvious. Additionally, some of the servicer requirements became design constraints on the spacecraft. Table 3 is a brief summary of the serviceable spacecraft analysis results and the servicer system requirements as seen by Martin Marietta. The rationale for the statements was given in the various IOSS presentations or can be readily demonstrated as being desirable.

A tier of modules is a layer of modules generally in a common plane and arranged so that all modules in a tier can be exchanged axially or all radially. While this recommendation, and the related recommendation on removal direction, are suggested to simplify design and operations, they are not constraints. The servicer system can handle a mix of radially and axially oriented modules. With regard to the interface mechanisms, which are the structural connections between the module and the spacecraft, it appears

Table 3 Serviceable Spacecraft Configuration Implications

- DOCKING SYSTEM
  - CENTRAL
  - NORMAL TO SOLAR ARRAY DRIVE AXIS
  - MINIMIZE DOCKINGS PER SERVICE
- SHAPE AND STRUCTURE
  - MAXIMUM OF TWO TIERS OF MODULES PER DOCKING
  - USE AVAILABLE ORBITER CARGO BAY DIAMETER
  - CONFIGURE FOR MINIMUM WEIGHT
- MODULES
  - SERVICE BOTH SUBSYSTEM AND MISSION EQUIPMENT
  - REMOVAL DIRECTION - AXIAL OR RADIAL, NOT BOTH
  - NUMBER OF MODULES - 10 TO 30
  - MODULE SIZE - 15 IN. CUBE TO 40 IN. CUBE
  - MODULE WEIGHT - 10 TO 700 LBS
  - HAVE STANDARD LOCATIONS FOR SUBSYSTEM MODULES
- INTERFACE MECHANISMS
  - STANDARDIZE INTERFACE WITH SERVICER AND STOWAGE RACK
  - AVOID THERMAL CONNECTORS REQUIRING CONDUCTION

desirable to permit the spacecraft designer to select his own configuration if he chooses. The only constraints are that it interface properly with the servicer mechanism end effector and the stowage rack.

As part of the serviceable spacecraft design work, the incremental costs of designing spacecraft were estimated. Each of the team members prepared estimates and the literature was also reviewed for incremental cost estimates. The consensus result was that design and development costs would be increased by 8% and unit costs would be increased by 4% of the non-serviceable spacecraft costs.

Additional Technology Applications

As an example of the adaptability of the selected on-orbit servicer system, various alternative servicing methods for a Multi-mission Modular Spacecraft (MMS) were analyzed. The recommended method for remote, on-orbit servicing on an MMS, such as the Solar Max Mission Spacecraft, was to use the standard servicer configuration fitted with a straight docking probe adapter, a modified Module Servicing Tool (MST) and a modified stowage rack (as shown in Figure 9). The servicer docks with the MMS laterally, on its existing grapple fixture or on a grapple fixture/berthing pin combination that replaces an existing berthing pin. An orientation joint, similar in design to the other servicer joints, is included in the docking probe adapter to allow tilting of the servicer with respect to the MMS after docking to bring the servicer mechanism into a plane parallel to the face of the module to be exchanged. The joint is powered through an electrical connection across the servicer docking interface. This feature allows the simple, axial mode of operation of the servicer without modifying its basic configuration. Either one of the two modules adjacent to the grapple fixture can be serviced in one docking. No modifications of the MMS modules or module retention system (MRS) are required. Instead, a modified MST compatible with the existing MRS and with the servicer standard end effector interface was recommended.

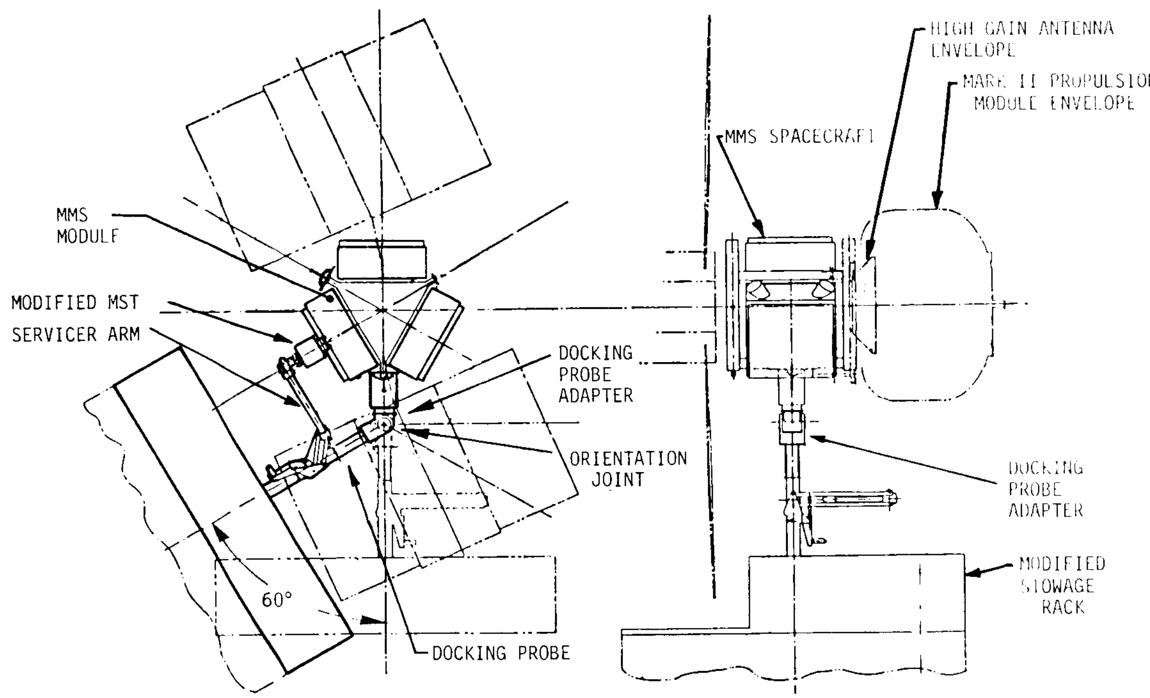


Figure 9 Servicing a Multi-Mission Modular Spacecraft

The IOSS functions of module exchange and umbilical connection for electrical signal or fluid transfer are widely applicable to the Space Station as shown in Figure 10. The sketch on the left hand side of the figure is an early Martin Marietta concept for servicing of objects that are brought to the Space Station. Examples of servicing functions that can be performed by the IOSS are listed on the right.

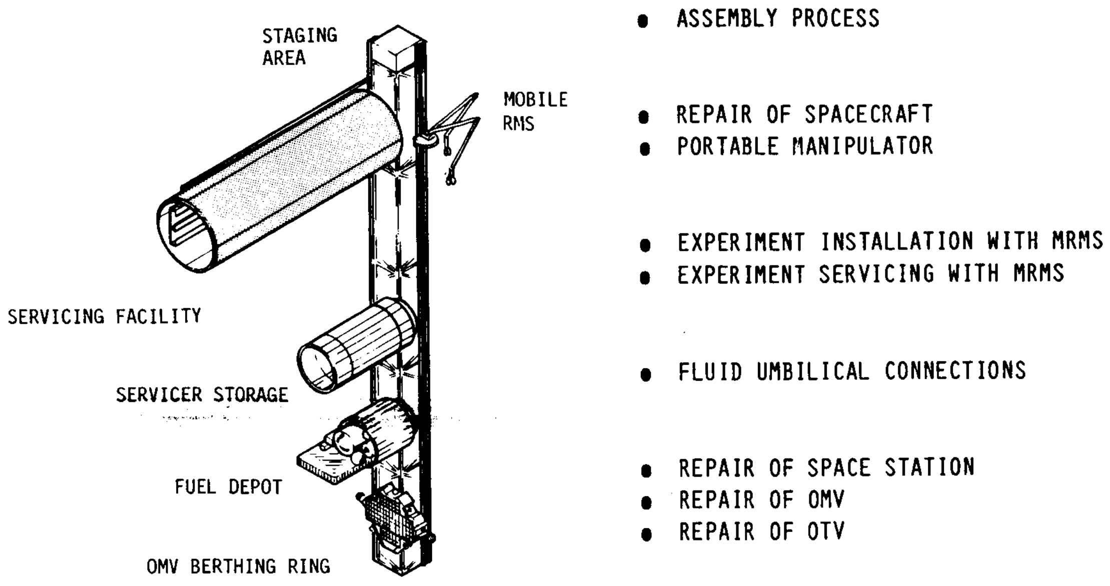


Figure 10 On-Orbit Servicer System At Space Station

The IOSS could be involved in the assembly process by bringing modules to prepared locations on the deployed trusswork. The prepared locations would also make it easy to replace any subsystems that subsequently fail. Experiments located on the Space Station framework far from the habitation modules could be installed and replaced when necessary using the servicer with the mobile RMS. The IOSS umbilical connection capability could fulfill the need to resupply both the OMV and the OTV. Another possibility, is to incorporate the IOSS concepts into the warehouses that store replacement modules much as trucks and fork-lifts are used in terrestrial warehouses. These and similar concepts could be used to reduce EVA workloads, especially those that are repetitive or hazardous. The intent of the ideas shown on the figure is more to outline possibilities and to open up alternatives, rather than to indicate recommended solutions.

The major conclusions of this work are:

1) The benefits of designing spacecraft for servicing are large compared to the costs;
2) Spacecraft design can greatly simplify the on-orbit servicer system;
3) The serviceable spacecraft design technology can be directly applied to the Space Station.

GAMMA RAY OBSERVATORY ON-ORBIT SERVICING

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ABSTRACT

The feasibility of performing on-orbit servicing of the NASA Goddard Space Flight Center (GSFC) Gamma Ray Observatory (GRO) was initially addressed by TRW during the later portion of the Phase C development contract in 1981/82. At that time, the investigation was specifically task limited to the potential for on-orbit replacement of selected mission-critical subsystems or components. Because of the advanced state of the design of the GRO scientific instruments at that time, any consideration for on-orbit servicing/replacement of the instruments was not addressed. Feasibility and concept definition tasks associated with the on-orbit refueling of GRO were addressed when the GRO Phase D contract was awarded early to TRW in 1983.

The GRO program completed its Preliminary Design Review (PDR) in May 1984, and Critical Design Review (CDR) in June 1985. The current GRO design reflects a capability for on-orbit changeout of the two Multimission Modular Spacecraft (MMS) modular power system (MPS) modules and the MMS communications and data handling (CADH) module via EVA. A sketch showing the GRO in a repair mission simulation, berthed to the FSS A-prime cradle, is shown in Figure 1. In addition, the design incorporates a capability for on-orbit refueling that is compatible with the JSC/Fairchild-developed, EVA-operated refueling coupler and the JSC Orbital Spacecraft Consumables Resupply System (OSCRS). The GRO design also incorporates a capability of EVA override operations for the deployment, restowage, and jettison of the GRO solar array and high-gain antenna appendages, the grapple fixture, and the electrical umbilical interface. A sketch of GRO in a deployment mission configuration prior to appendage deployment is shown in Figure 2.

To validate the GRO EVA design compatibility prior to CDR, a series of five separate astronaut-suited test runs were performed at the NASA/JSC Weightless Environment Training Facility (WETF) using a high-fidelity full-scale mock-up (FSM) of the GRO. These EVA simulation tests to evaluate the

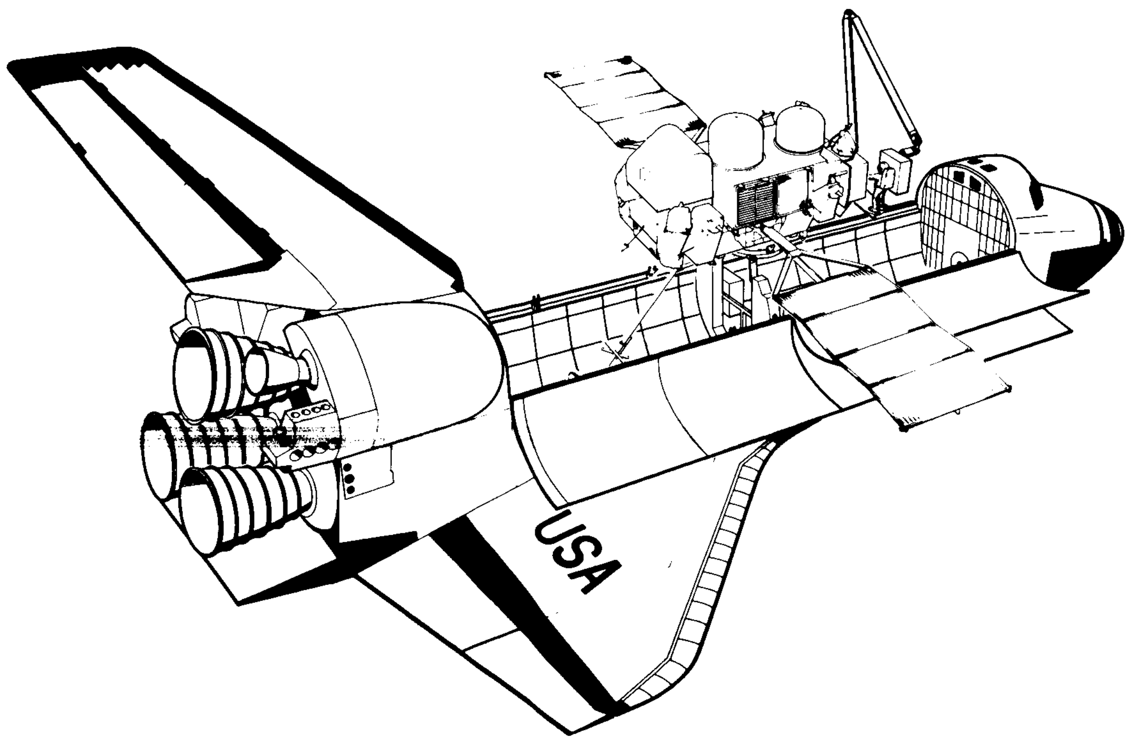

Figure 1. Refueling/Repair Mission Configuration: GRO on A-Prime Cradle

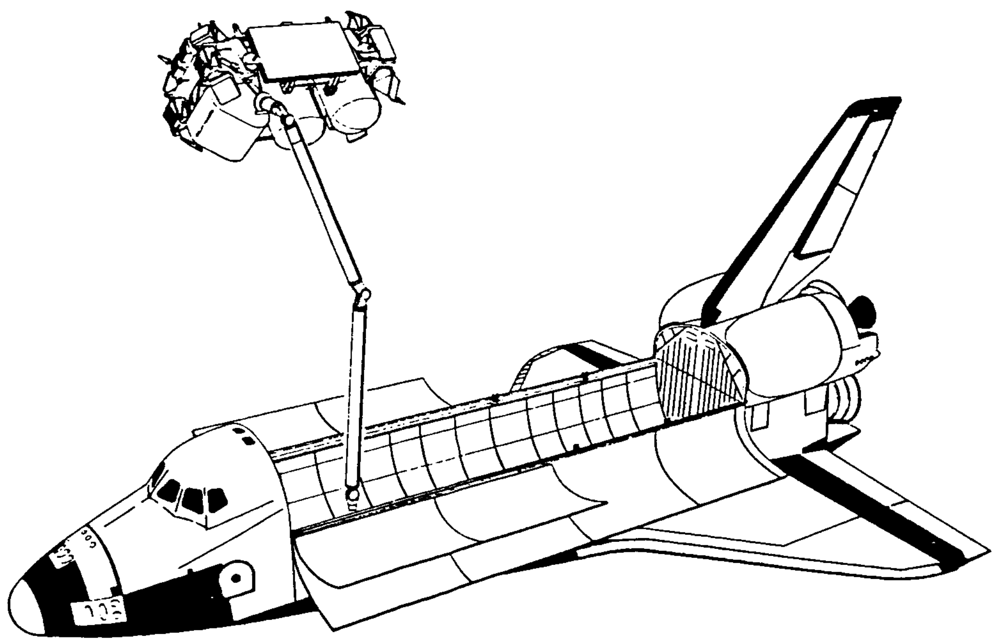

Figure 2. GRO Deployment on RMS

GRO on-orbit servicing compatibility were supported by five astronauts, all with prior flight EVA experience. The tests were performed from 5 March to 3 April 1985.

INTRODUCTION

The studies performed by TRW to determine the feasibility of on-orbit servicing, repair, and refueling were performed under a very specific set of NASA assumptions and ground rules. The GSFC GRO Project Office was (and is) under heavy pressure to maintain program costs and schedule commitments established before the on-orbit servicing discussions were initiated. NASA headquarters initiated the first request to the GSFC GRO Project Office to investigate on-orbit servicing for GRO. No additional funding was provided, however, to support the feasibility studies or the subsequent design and implementation efforts.

The selection of the MMS power and communications modules for incorporation into the original baseline design was recommended by TRW during the later portion of the GRO Phase B concept definition contract. This selection was recommended principally as a program cost savings, i.e., to use an existing, qualified, flight-proven design. The attendant on-orbit replacement capability of these modules was not, at that time, considered to be a significant advantage to the GSFC GRO project. The conceptual design of GRO during the majority of the Phase C design definition phase did not include any provision for astronaut EVA involvement in either a planned or contingency support operational role. As an amendment to the Phase D RFP, TRW was asked to identify design modifications and costs associated with incorporating an on-orbit EVA module changeout capability for the power and communications modules, and an EVA-supported appendage deployment manual operation as a contingency should the automatic deployment system fail to operate. In addition, the amendment to the RFP asked to identify the design and cost impacts for making GRO retrievable by the orbiter. These initial maintenance EVA override, module replacement, and orbiter retrievability features were incorporated into the Phase D contract Statement of Work to TRW in February of 1983. Commensurate with the start of the Phase D contract, the GSFC GRO Project Office directed TRW to perform a feasibility and concept definition study to establish technical, cost, and schedule impact for incorporating an on-orbit refueling capability for GRO. The study was completed in 90 days, and the GRO Phase D contract was modified in June 1983 to incorporate an on-orbit refueling capability into the baseline GRO design.

GRO DESCRIPTION

A summary of the GRO program milestone is shown in Table 1. Figure 3 is a summary GRO project schedule. The GRO mission objectives are summarized in Table 2 and the overall mission concept is depicted in Figure 4.

Table 1. GRO Program Summary

Sponsor: NASA (Office of Space Science)

Customer: NASA Goddard Space Flight Center

Mission contractor: TRW

Program chronology:

- Mission need statement issued in May 1978
- Phase 1 studies conducted in 1980
- Program approval document issued in February 1981
- Phase C contract from April 1981 through September 1982
- Phase D contract from February 1983 through mission end
- PDR in May 1984
- NASA/JSC WETF testing February to April 1985
- CDR in June 1985
- Launch in May 1988. Inclination 28.5 degrees; mission altitude 350 to 450 km
- Two-year science mission
- STS retrieval return from orbit (1990+)

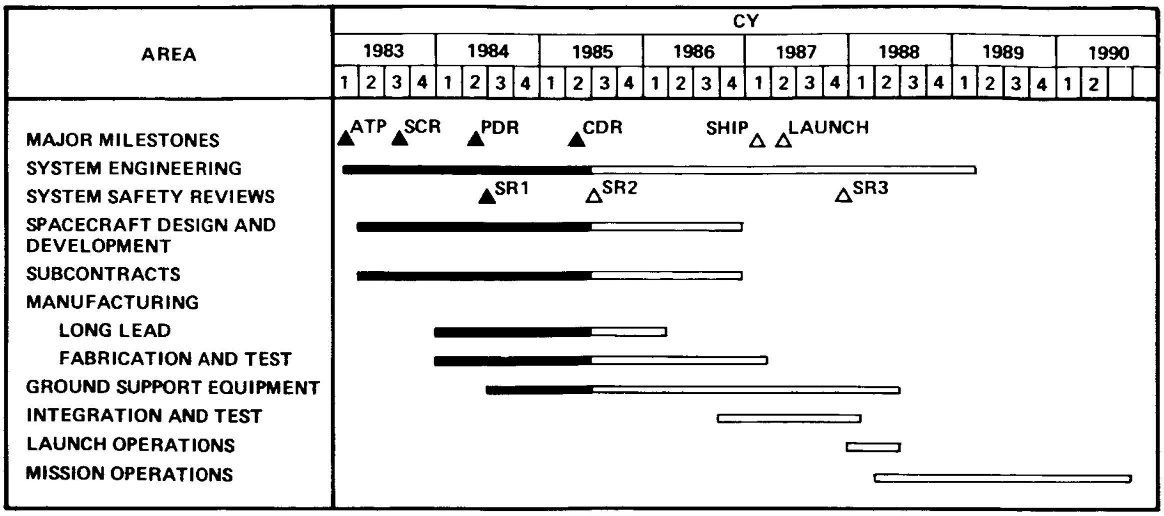

Figure 3. GRO Project Schedule

Table 2. Mission Objectives

Study dynamic evolutionary forces in compact objects such as neutron stars and black holes
Search for evidence of nucleosynthesis
Investigate gamma-ray-emitting objects whose nature is not understood
Explore our galaxy in the gamma-ray range, particularly with regard to regions difficult to observe at other wavelengths
Study the nature of other galaxies in the energetic realm of gamma rays
Study cosmological effects through detailed examination of the diffuse radiation and the search for primordial black hole emission

GRO On-Orbit Serviceability

As previously mentioned, program cost considerations significantly limited detailed investigations and conceptual design efforts to establish additional on-orbit servicing capabilities, e.g., component/module or subsystem changeout. The GRO design proposed for the Phase D development contract incorporated extensive use of qualified, flight-proven hardware that was in many instances not readily modifiable to an ORU configuration. A subsystem component reliability analysis was performed on the GRO

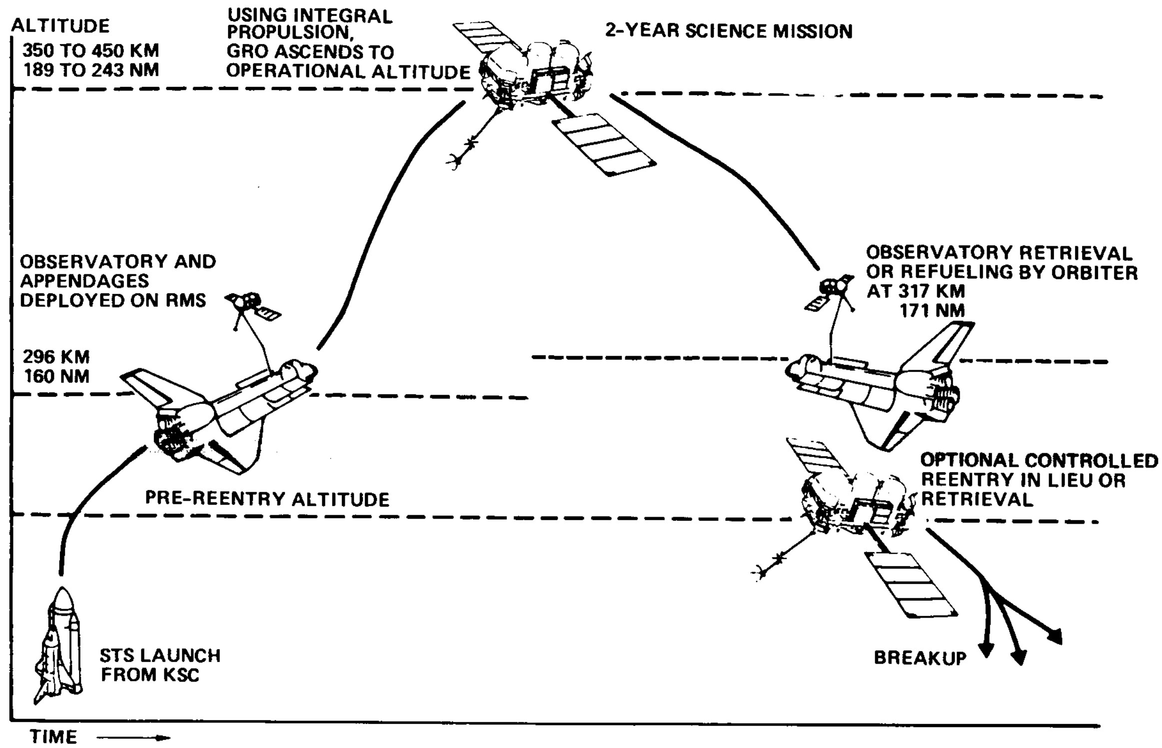

Figure 4. GRO Mission Concept

attitude control and determination (ACAD) subsystem to establish data on mean time before failure rates and determine what components, if any, should be considered as candidates for on-orbit replacement. Considering the 2-year nominal mission lifetime, this analysis showed that no single component within the ACAD subsystem should be designed for on-orbit replacement, and the overall ACAD subsystem reliability numbers supported the same conclusion. At the start of the Phase D contract, the two MMS power modules and the MMS CADH modules were baselined as the only GRO on-orbit replaceable components/subsystems.

Deployment Mission – Initial Maintenance

The studies performed near the completion of the Phase C contract suggested that an improvement in mission reliability could be achieved if certain mission-critical automatic appendage deployment functions could incorporate an EVA override feature. As part of this effort, a motor-driven appendage release and deployment system was incorporated in place of the original ordnance-activated, spring-release system. Manual EVA wrench-actuated overrides were incorporated into the gear drives of the motor-driven appendage release and deployment mechanisms on the two solar arrays and high-gain antenna booms.

Deployment/Retrieval Mission EVA Evaluation

The anticipated planned and contingency EVA operations for both the GRO deployment and retrieval missions are similar. If a solar array or high-gain antenna appendage mechanism fails to perform satisfactorily, an astronaut in EVA, using standard wrenches and tethers, can override the electrical drive motor and deploy or restow the affected appendage.

To perform most of the EVA operations that may be required on the GRO deployment mission, the EVA test crew will not have the RMS/manipulator foot restraint (MFR) available; the RMS is being used in conjunction with the GRO grapple fixture to hold the GRO above the open cargo bay. All of the EVA operations associated with solar array or high-gain antenna appendage latch release and deployment must be performed using the portable foot restraint (PFR) units presently in the orbiter inventory. An EVA operational flow is shown in Figure 5.

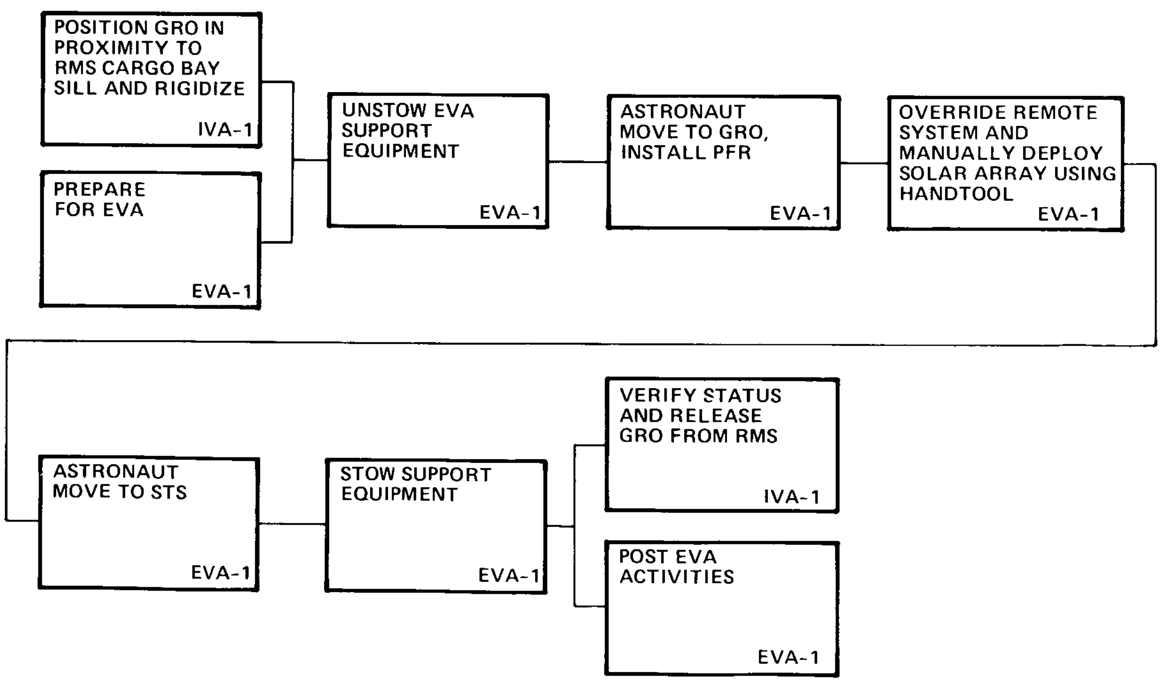

Figure 5. GRO Deployment Mission EVA Flow Chart

These EVA override operations using the PFRs were rehearsed as part of the GRO FSM WETF activities. As a result of these tests, a change was incorporated into the flight design of the GRO solar array appendage to provide improved access to the array jettison bolts. In addition, the crew personnel recommended that additional handrails and foot restraint sockets be added to improve EVA accessibility. This hardware has been incorporated into the flight design.

Repair/Refueling Mission EVA Operations

The GRO design incorporates the capability for on-orbit replacement of either of two power modules and/or the communications and data handling (CADH) modules. The mechanical design of these modules is identical to that of the GSFC-developed MMS modules previously flown on the Solar Max and Landsat missions. A sketch of the module is shown in Figure 6. The

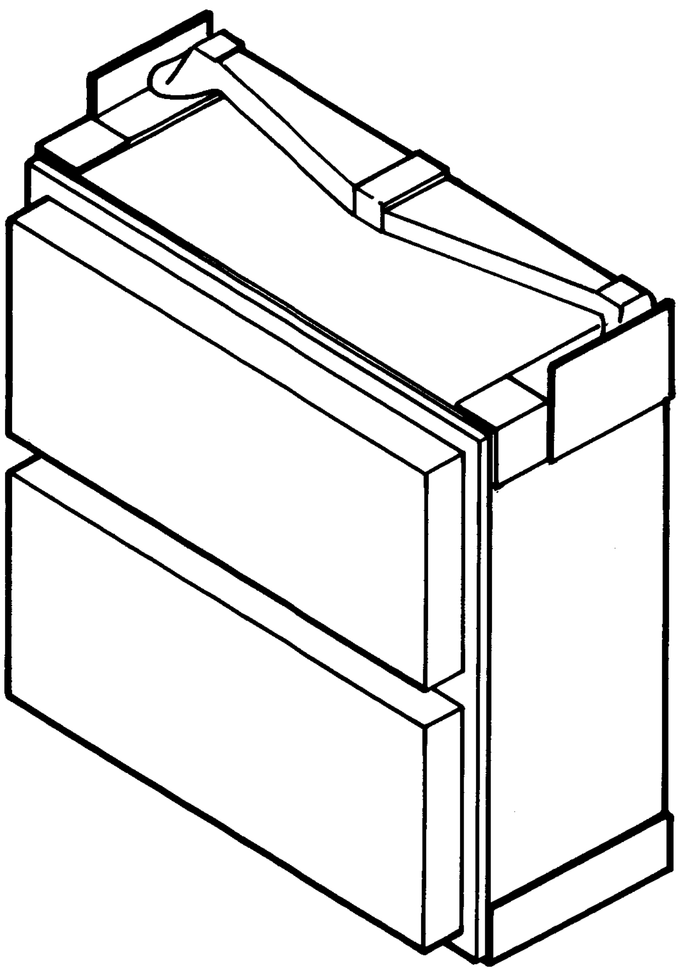

Figure 6. MMS MPS and CADH Module Configuration

recent successful STS/Solar Max repair mission validates the on-orbit changeout capability of this package design as well as the support tools and hardware/software used in the operation.

To assure GRO/orbiter refueling interface compatibility, GRO personnel have maintained close communications with the NASA/JSC propulsion branch personnel within the engineering directorate. This task involved the review/critique of the original requirements and SOW documentation for both the on-orbit refueling coupling (payload/orbiter propellant interface) and the OSCRS. The coupling is currently in final development by Fairchild Controls, and the OSCRS program is currently in an 8-month preliminary design study with five contractor teams participating. The current coupling design reflects design improvements that were incorporated as a result of the GRO OOR EVA WETF evaluation testing performed early in 1985 and repeated in June of 1985. The OSCRS RFP/SOW specifically addresses the requirement for compatibility with the GRO propulsion subsystem. JSC is currently planning on the initial OSCRS development and operational readiness by 1990.

Repair/Refueling Mission EVA Evaluation

Of primary concern in the EVA box changeout operations simulated in the WETF testing was the establishment of crew translation routes between the box location on GRO and the box storage location on the FSS A-prime cradle used to berth the GRO during these operations (Figure 2). A mock-up of the on-orbit refueling coupling had been installed on the GRO structure prior to the start of the FSM WETF activities. One of the EVA tests was devoted to establishing the preferred position for the astronauts during the refueling coupling mate and demate operations. This test was also performed with GRO berthed to the FSS A-prime cradle in the actual mission simulation configuration. Specific astronaut recommendations for EVA design enhancement for support of GRO repair/refueling mission operations included the addition of handrails and portable foot restraint sockets in specific locations and a requirement for an EVA-installed handling fixture for moving the modules between the worksite locations. In addition, an area of interference between the GRO integral berthing adapter structure and the FSS A-prime cradle latch motor case was identified. The GRO structure will be modified to eliminate this interference.

GRO/STS Interfaces

Table 3 provides a summary of the GRO/STS interfaces for each of the three missions. Figure 7 graphically identifies these interfaces.

Table 3. GRO/STS Interface Summary

INTERFACES	DEPLOYMENT	REPAIR/ REFUELING	RETRIEVAL
1. STANDARD FIVE-POINT ACTIVE TRUNNION INTERFACE	X		X
2. DEPLOYMENT, BERTHING, AND RESTOW USING RMS/GRAPPLE FIXTURE STANDARD INTERFACE	X	X	X
3. BERTHING TO FSS A PRIME CRADLE VIA GRO INTEGRAL BERTHING ADAPTER		X	
4. ELECTRICAL POWER AND HEATER CONTROL THROUGH AESE	X	X	X
5. STANDARD UMBILICAL RELEASE SYSTEM (SURS) POWER AND SIGNAL INTERFACE	X		X
6. PF1 MDM INTERFACE DURING IN-BAY POWER-OFF OPERATIONS	X	X	X
7. PI/PDI INTERFACE FROM CADH TO TDRSS/MCC/POCC VIA LGA	X	X	X
8. AFT FLIGHT DECK (AFD) STANDARD SWITCH PANEL (SSP) FOR GRO POWER CONTROL AND SAFETY STATUS MONITORING	X	X	X
9. FHST SHUTTER CONTROL FROM SSP	X	X	X
10. AUXILIARY EVA UMBILICAL FOR MONITORING OF CRITICAL OOR PARAMETERS		X	
11. PLANNED AND UNSCHEDULED EVA			
• APPENDAGE DEPLOYMENT/ RESTOW/JETTISON	X		X
• ORU (MPS, CADH) CHANGEOUT		X	
• REFUELING		X	

GRO FOLLOW-ON SERVICING POTENTIAL

When the GRO has completed its scientific mission, it could be used as a spacecraft on which to conduct technology demonstration and crew training to advance on-orbit satellite servicing. With the Space Station as the host vehicle, a series of servicing technology development missions (TDM)

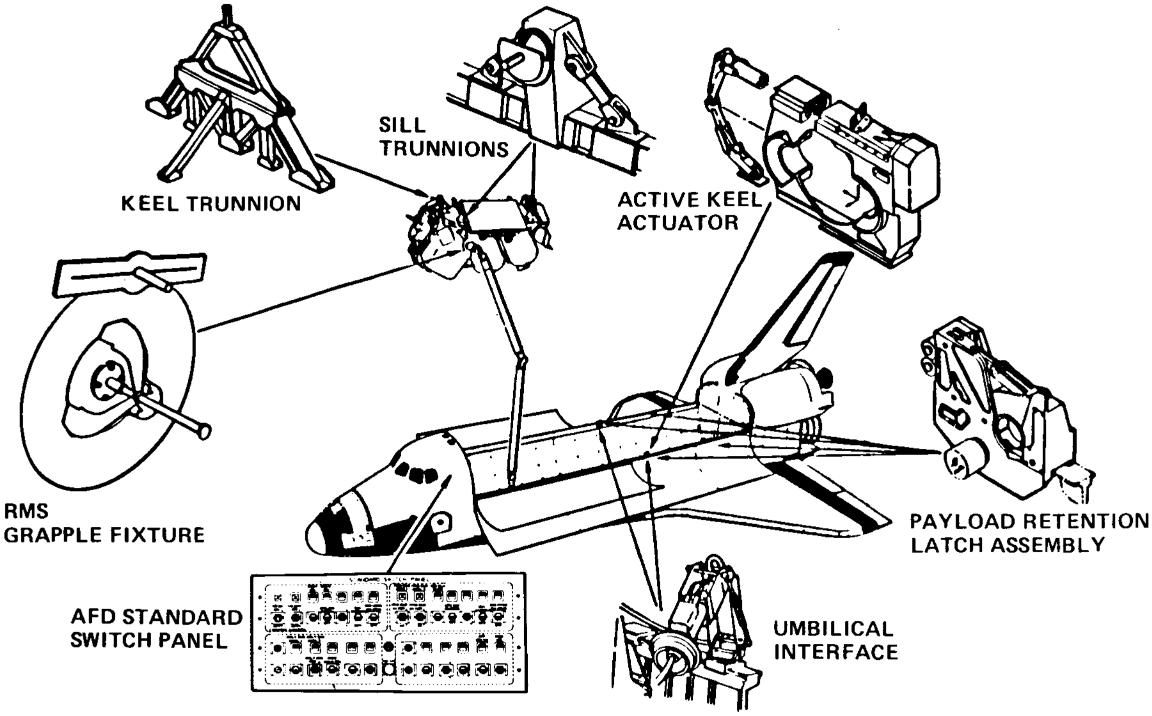

Figure 7. GRO Electrical/Mechanical Interfaces with Orbiter

is envisioned. The technology considerations, benefits, Space Station requirements summary, and scenario highlights are listed below:

1) Technology considerations

- EVA construction/disassembly
- On-orbit fluid transfer/storage
- OMV operations
- Part replacement
- Contingency service operations
- On-orbit system/subsystem test
- Satellite retrieval
- Advanced crew support technologies

2) Benefits

- Extension of life of GRO
- Applicable to repair/refurbishment of many other spacecraft

3) Space Station requirements

- Mechanical and electrical support equipment
- Crew support equipment
- Refillable propellant tanks
- Special crew training
- Autonomous mission support systems

4) Scenario highlights

- GRO retrieval from 400 km orbit
- Comprehensive status tests
- Refurbishment/repair of units
- Propellant refill
- Comprehensive checkout
- Redeployment into operational orbit.

TDM Description

The objective of the TDM is to demonstrate the capability to service a low earth orbiting satellite, in this case the GRO, at the Space Station. Such servicing will extend the useful life of the spacecraft. GRO was picked as an example.

Because of its great size, special arrangements must be made to service the GRO at the Space Station. It would be desirable to attach the GRO to the servicing shelter cargo rails with the "skin" of the shelter removed. This would permit the use of extended payload retention latch assemblies (PRLA) to allow adequate space for the refueling operation and access to orbital replacement units (ORU).

Sequence of Events

The Orbital Maneuvering Vehicle (OMV) "flies" out to rendezvous with the GRO, attaches to the grapple fixture (located above the trunnion mount), and maneuvers the spacecraft toward the Space Station. To facilitate this operation, the grapple fixture must be oriented toward the GRO center of gravity.

When the GRO is very near to the Space Station, the module manipulator system connects to the OMV grapple fixture. An astronaut in a manned maneuvering unit goes out and mounts a portable grapple fixture to the end of the GRO satellite. A handling and positioning aid (HPA) can be attached to this portable grapple fixture to secure the spacecraft while the OMV is demated from the permanent grapple fixture and stored using the module manipulator system. The combined capabilities of the module manipulator system and HPA can then be used to position the GRO against the PRLAs attached to the cargo rails. Attachment will be made remotely from inside the Space Station.

The next step is changeout of an orbital replacement unit. The type of unit to be replaced will be determined at the time the demonstration is planned, based on requirements to extend the life of the spacecraft. If solar arrays need to be replaced, the entire array, including its drive assembly, will be changed out. If no subsystems require replacement, a standard command and data handling module could be changed out to demonstrate the technique. ORU changeout will be performed by two suited astronauts using portable handholds and foot restraints, wing tab connectors, and the module manipulator system and HPA.

After the changeout, the astronauts will set up a fueling kit and position the fueling (and pressurizing) connector(s) against the fueling port and hold it (them) there with the HPA (and module manipulator system). The astronauts then return to the Space Station and the coupling of the fuel connector is completed remotely. This reduces the risk of space suit contamination, enchancing crew safety.

After refueling, fuel lines are evacuated and uncoupled from the spacecraft. Then the OMV is mated to the spacecraft, the portable grapple fixture is removed, and the GRO is returned to optimum low earth orbit.

Benefits and Applications

This TDM will demonstrate the capability to retrieve a LEO spacecraft, bring it to the Space Station, perform necessary servicing, and return it to the optimum orbit, thereby extending useful satellite life. This capability has applications to virtually all LEO satellites, and will enable more sophisticated servicing operations that can be performed by remote (in situ) operations or by servicing with the STS orbiter.

The increased capability enabled by satellite servicing at the Space Station provides the following benefits.

1) The spacecraft can be disassembled for access to connectors, sensors, and other equipment (the service platform provides room for storage and tie-down during servicing operations).

2) Large, complex components, such as solar arrays, can be replaced or refurbished and tested prior to spacecraft redeployment.

3) Spacecraft optical, thermal, and solar array surfaces can be cleaned or refurbished.

4) Large, fragile spacecraft (those assembled, tested, and inserted into orbit from the Space Station) can be serviced with reduced risk of damage.

In addition to the increased servicing capability, the following benefits can be realized.

1) The spacecraft capability can be upgraded by retrofit to provide, for example, more power from increased solar array area and/or more battery capacity, more accurate stationkeeping with improved sensors, and more reaction control capacity from added fuel capacity.

2) The spacecraft mission can be altered by replacing existing experiments or functions with others.

3) The spacecraft orbit can be changed with appropriate sensor changes and reinsertion into the new desired orbit via the OMV.

Special Considerations

This TDM is baselined using the GRO as the service object. While the GRO is being designed for limited on-orbit servicing via the STS orbiter, several special considerations are applicable for Space Station servicing:

1) The grapple fixture (used to remove the GRO from the orbiter payload bay) must be oriented toward the center of gravity of the spacecraft to permit retrieval and reboost by the OMV.

2) The GRO design must include provision for attaching a second (portable) grapple fixture for handling at the Space Station.

3) The GRO refueling equipment to be used by the orbiter must be compatible with Space Station capabilities.

DESIGN EXPERIENCES IN DEVELOPMENT OF EVA SERVICEABLE INSTRUMENTS FOR THE HUBBLE SPACE TELESCOPE

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ABSTRACT

The Science Instruments and Fine Guidance Sensors of the Hubble Space Telescope are EVA serviceable at the module level. Precision alignment of these EVA replaceable modules is critical to system performance. Development of the mechanisms (registration fittings) to accomplish repeatable alignment is a significant accomplishment. The design requirements, features and realized performance of these registration fittings are presented in this paper.

INTRODUCTION

The Hubble Space Telescope is comprised of a Ritchey-Chretien Cassegrain Telescope and the five Science Instruments and three Fine Guidance Sensors that simultaneously share its partitioned aplanatic focal surface. Each of the Science Instruments and Fine Guidance Sensors, referred to as instruments, is EVA replaceable. Since each instrument must be precisely located with respect to the telescope focal surface, a mechanism for facilitating replaceability while maintaining precision positioning had to be developed.

As finally configured, each instrument is supported by three, or four registration fitting pairs. Each fitting pair serves to constrain the instrument in one or more translational axes; in aggregate the fittings constrain each instrument in six degrees of freedom with respect to the telescope structure.

EVA removal of each instrument is accomplished by release and translation of the instrument out of the telescope through doors in the spacecraft skin. Since each instrument is quite large, weighing 2224 to 3114 Kg's (500 to 700 pounds) and having a maximum dimension of 1.5 to 2.1 meters (5 to 7 feet), guide rails are provided to assist the crewmen in controlling the unit. Replacement is accomplished by reversing the process. The general arrangement of the instrument and the telescope is shown in Figure 1.

DESIGN REQUIREMENTS

The key design requirements associated with EVA replaceability are position repeatability and crew systems compatability. Satisfactory alignment of the telescope instruments requires control of many error sources including initial alignment, residual launch deformations, thermally induced distortions, tooling and measurement errors and, in the case of EVA replaceable instruments, position (alignment) repeatability.

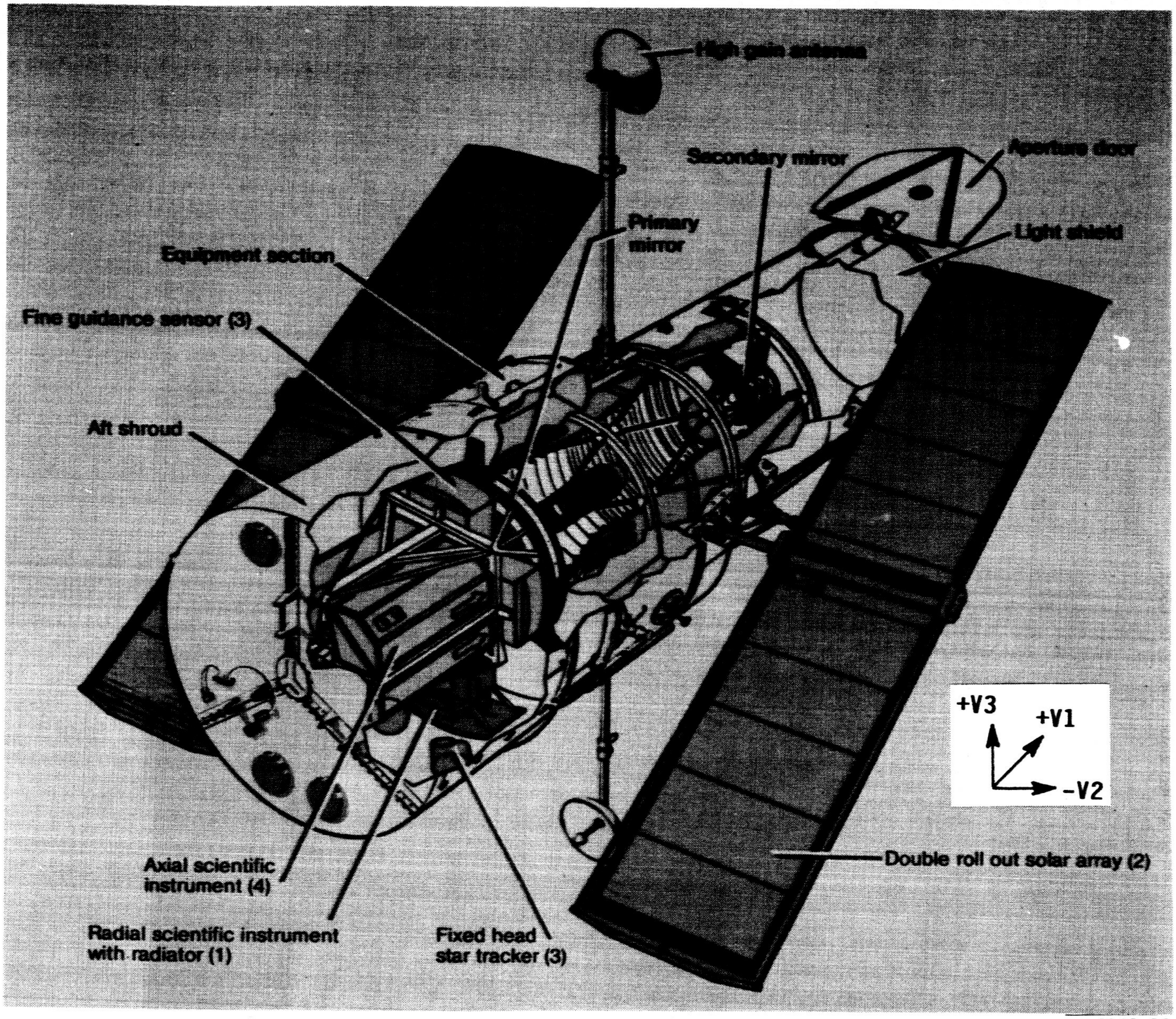

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Figure 1. Hubble Space Telescope configuration

Total position error for an optical instrument is traditionally presented as being comprised of despace, decenter, and tilt components. Despace is a measure of position error along the telescope optical axis and directly corresponds to errors in focus. Decenter is a measure of position errors orthogonal to the optical axis. Tilt is a measure of angular alignment deviations relative to the telescope optical axis.

It was necessary to limit the magnitude of the axial and radial science instrument position repeatability error to approximately 25% of the maximum allowable position (alignment) error requirement as set forth in the telescope alignment error budget. The Fine Guidance Sensor contains a wavefront sensor to assess the telescope wavefront quality. Wavefront sensor performance requirements dictate closer overall control of positioning of the Fine Guidance Sensor instruments and repeatability error contributions were allocated a larger portion (approximately 75%) of the maximum allowable position error requirements.

It should be noted that over the course of the program, considerable effort was expended to control other error contributors to allow a twofold increase in the repeatability error allocation.

MAXIMUM ALLOWABLE INSTRUMENT POSITION (ALIGNMENT)
DUE TO ALL SOURCES

INSTRUMENT	DESPACE (μ m)	DECENTER (μ m)	TILT (arc sec)
Axial Scientific Instrument(s)	153 (0.006")	153 (0.006")	30
Radial Scientific Instrument	254 (.010")	102 (.004")	74
Fine Guidance Sensor(s)	51	64	10 Tangential Axis
	(0.0020")	(0.0025")	20 Radial Axis

Note: Unless otherwise stated these decenter and tilt error budgets, as specified, are applicable to each of the two applicable orthogonal components separately.

The crew systems compatability requirement dictated simplifying the actions required by the crewmember to disengage/engage each instrument from the telescope. It was desirable, if not mandatory, to provide guide rails and a common interface for crewmember actuation of instrument retention mechanization.

Other design requirements that significantly influenced the resulting design solution are launch loads, stiffness, alignment thermal stability, thermal conductivity and residual moment constraints. The following tabulation presents the nominal range of requirements for the registration fitting pairs.

CHARACTERISTIC	NOMINAL RANGE OF REQUIREMENTS
Launch load induced **RESULTANT FORCE** for a registration fitting pair	13,300 - 22,200 N (3,000 - 5,000 lb.)
STIFFNESS of a registration fitting pair	20.5×10^6 to 42.9×10^6 N/m (117,000 to 245,000 lb/in)
ALIGNMENT THERMAL STABILITY contribution of an instrument complement of registration fittings	0.0013 arc sec over 24 hours
THERMAL CONDUCTIVITY	.05 to .08 w/°C maximum
RESIDUAL MOMENT for a fitting pair	31 to 54 NM Maximum (23 to 40 ft-lb)

In aggregate, these requirements force the designer to make numerous design trades in reaching an acceptable design solution.

DESIGN DESCRIPTION

The registration fitting complement for each instrument is designed to nominally provide a kinematic mount in six degrees of freedom. Except for comparatively small residual friction and/or preload mechanization torques, moments are not carried across a fitting pair.

Each of the registration fitting pairs embodies a ball-in-socket design which provides a self-alignment capability, and insures a nominally statically determinate interface with low-moment load transfer. A functionally similar design, incorporating flexures in lieu of the ball-in-socket, was considered, but envelope, strength, stiffness and thermal conductivity constraints precluded such a design solution.

AXIAL INSTRUMENT REGISTRATION FITTINGS

The instruments known as Axial Science Instruments are each supported in the telescope by three registration fitting pairs (Figure 2). The "A" fitting pair (Figure 3) restrains the instrument in three degrees of freedom. A 44.5 mm (1.75 in) diameter ball is mounted to the instrument and its mating spherical seat is mounted to the telescope structure. The spherical seat is segmented and mechanized to allow opening by the crewmember via a screwdrive to allow acceptance of the ball and subsequent closing to capture it. Ball-to-seat fit is maintained at 1.9 to 3.8μm (75 to 150 μin) so as to insure rotational freedom without producing large amounts of free play. Ball-to-seat radial fits of other registration fittings are similarly toleranced. In the launch environment excessive free play could produce significant shock loads at the interface. The ball and seat are 440C stainless steel and tungsten carbide/cobalt coated titanium respectively. Lubricated with Braycote 3L38RP, this material combination assures low friction torques and high galling resistance.

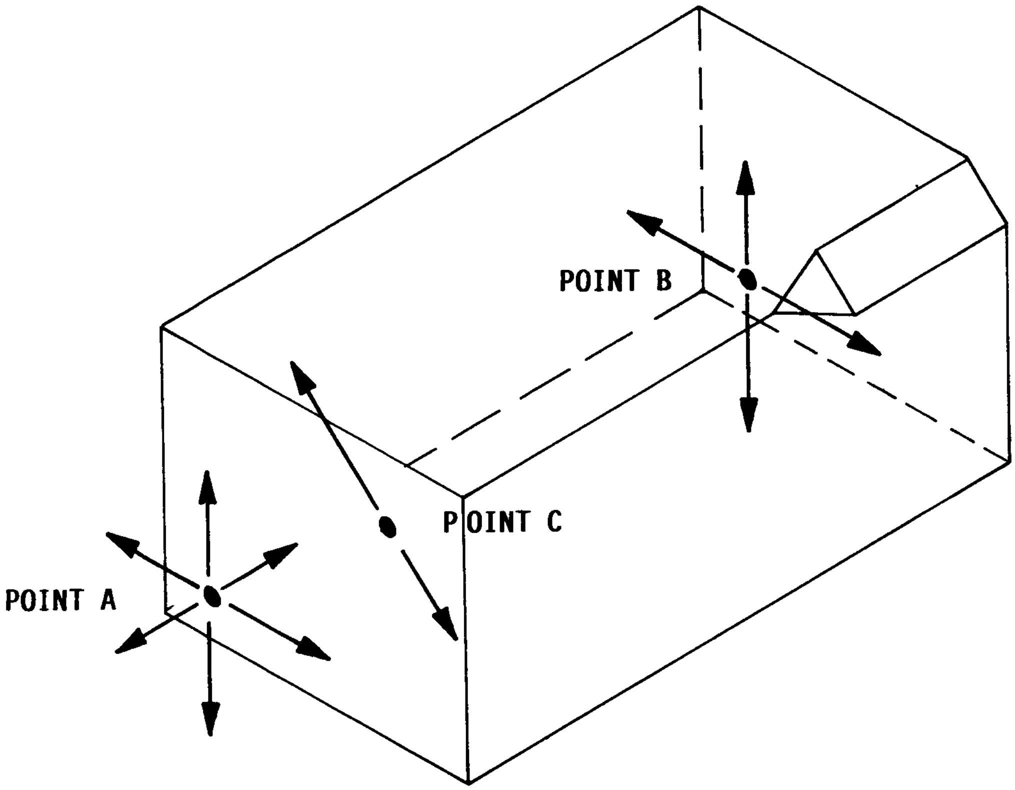

Figure 2. Axial science instrument showing registration fitting constraint directions

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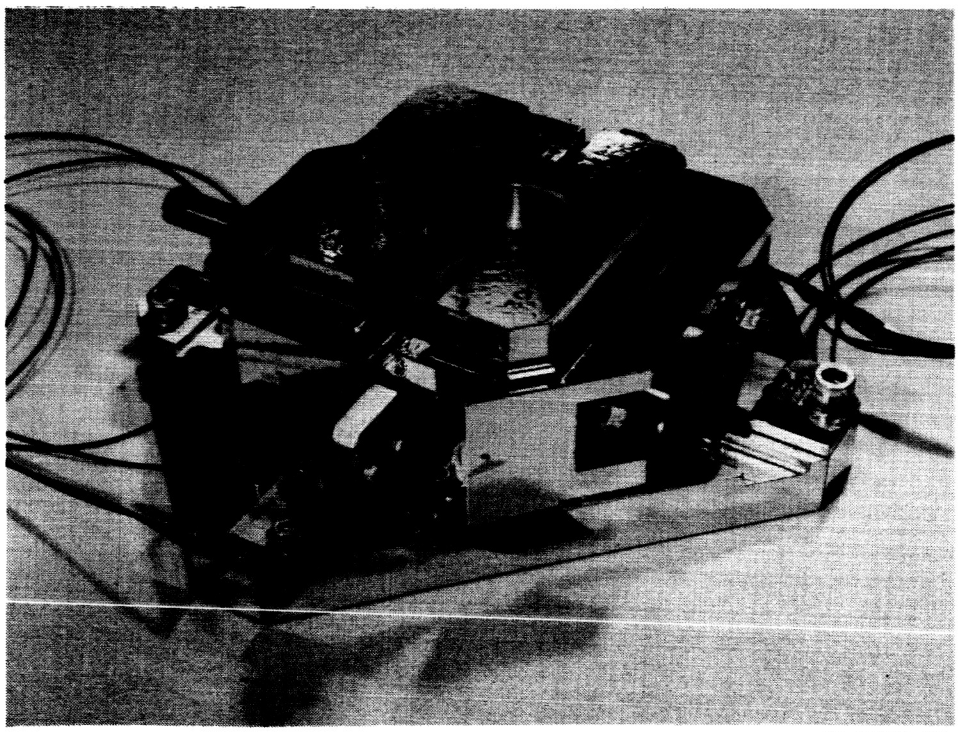

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Figure 3. Axial instrument "A" registration fitting pair

The "B" registration fitting pair (Figure 4) is mounted at the other end of each Axial Instrument and consists of a fully captured ball-in-socket mounted to the instrument and a spring-loaded restraining pin mounted to the telescope structure. The "B" fitting pair restrains the instrument in two degrees of freedom.

The pin preloads the instrument into the "A" latch with 3,559 Newtons (800 lbs.) of force, thus maintaining positive registration in both ground test and orbital environments. High preloads tend to reduce registration repeatability errors but induce higher friction torques. The existing preload reflects a 27% reduction necessary to achieve residual moment requirements.

Lastly, the Axial Instrument "C" registration fitting pair (Figure 5) provides restraint in a single degree of freedom. A self-aligning cylinder is mounted to a ball structurally mounted to the instrument. A mating receptacle, consisting of a flat and a parallel flexure, is mounted to the telescope structure. The flexure is designed to provide a preload sufficient to insure proper registration during operational modes only. Flight loads are supported by snubbing the flexure 64 μm (0.0025 in) beyond its normal rest position. A tungsten carbide/cobalt coating is applied to the 15-5 PH steel flexure to prevent galling and the fitting pair interface is lubricated with Braycote 3L38RP. The fitting pair self-aligns and mates when the instrument is translated in the +V1 direction. It should be noted that this construction is typical of the "B", "C" and "D" fitting pairs that mount the Radial Science and Fine Guidance Instruments.

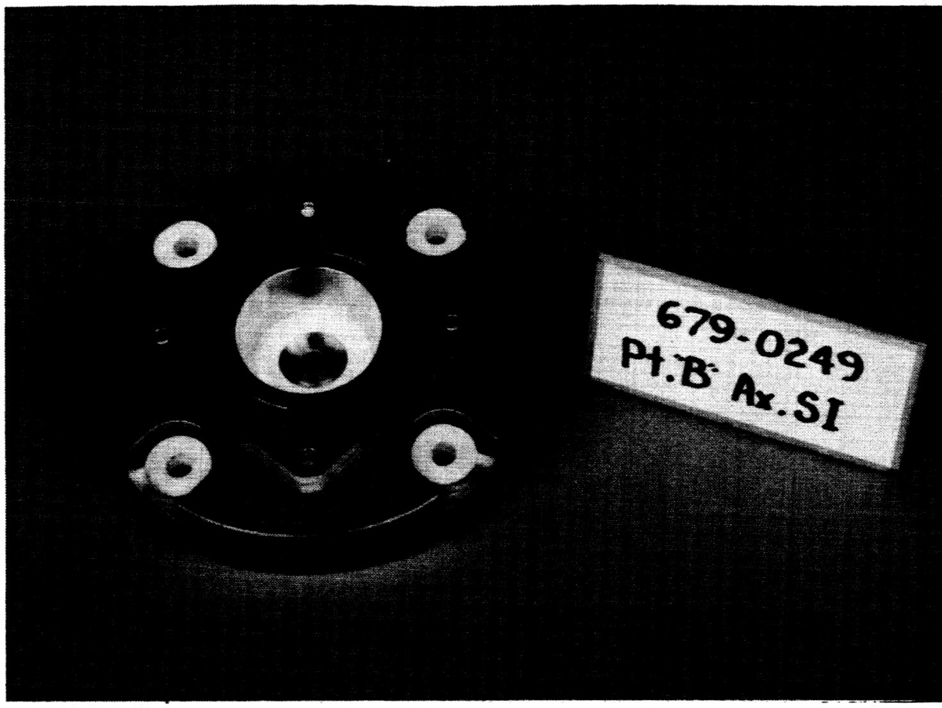
The Axial Instrument "A" and "B" registration fitting screw drives are separately actuated by the crewman with a torque-limiting socket wrench. Torque is transmitted from the drive point, at a convenient location on the telescope structure, to the fitting screw drive through drive rods with universals at each end. The "A" and "B" fitting drive torques are 60 NM (44 ft-lb) and 10 NM (7.5 ft-lb) respectively.

On-orbit removal of the Axial Instruments is accomplished by retracting the "A" and "B" Registration fittings followed by translation of the instrument in the -V1 direction and subsequent guided translation out through vehicle doors. Instrument installation is accomplished by reversing the process.

RADIAL INSTRUMENT REGISTRATION FITTINGS

The Radial Science Instrument and each Fine Guidance Sensor are supported in the telescope by three and four registration fitting pairs respectively. In each case the "A" fitting pair restrains the instrument in three degrees of freedom (Figure 6). A truncated 50.8 mm (2.000 in) diameter ball-in-socket is mounted to the telescope structure. The ball contains a threaded hole that serves as the fastening interface with the instrument-half of the registration fitting pair. A registration preload of 1245 N (280 lbs) is maintained at the "A" registration fitting via flexure springs that span between the fitting halves. The Radial Instrument "A" fitting pair is shown in Figure 7.

The Radial Science Instrument "B" registration fitting pair and the balance of the Fine Guidance Sensor registration fitting pairs are single axis constraint fittings similar to the Axial Instrument "C" fitting pair described earlier.



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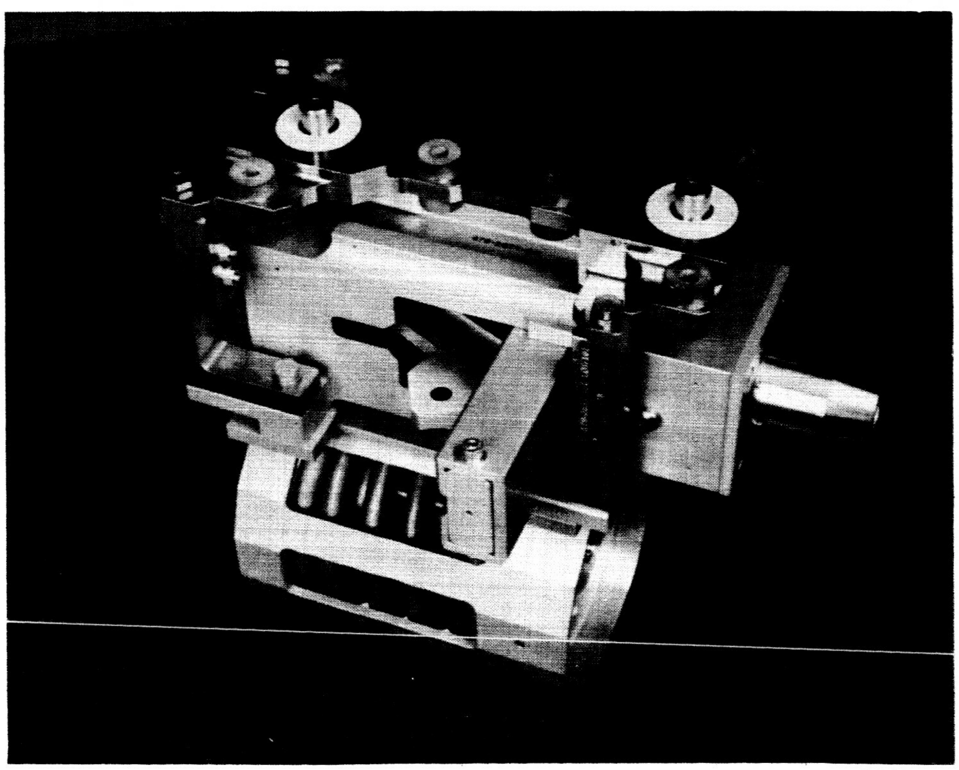

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Figure 4. Axial instrument "B" registration fitting pair

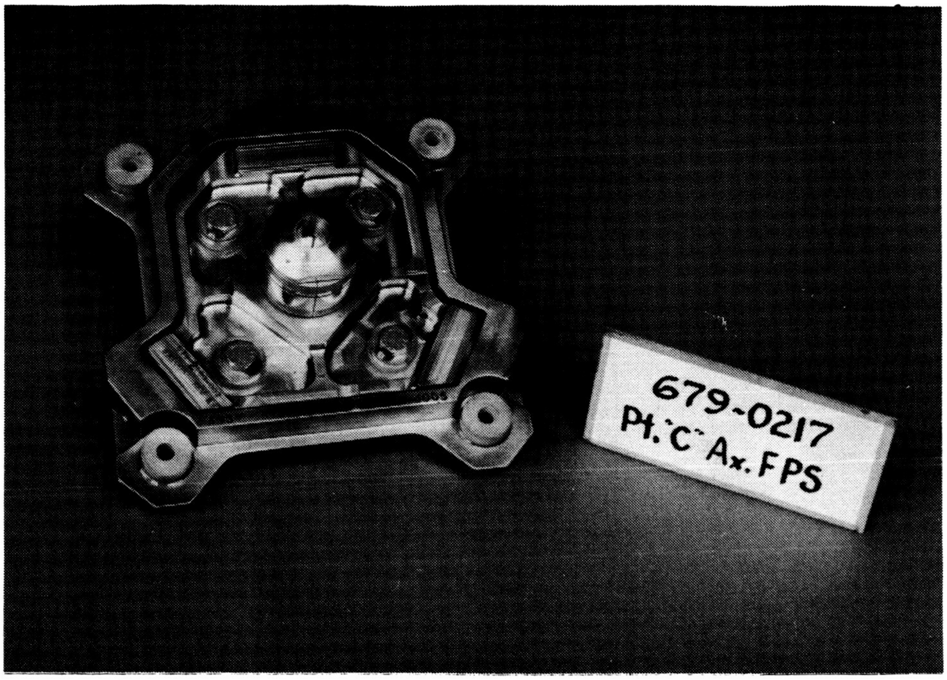

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Figure 5. Axial instrument "C" registration fitting pair

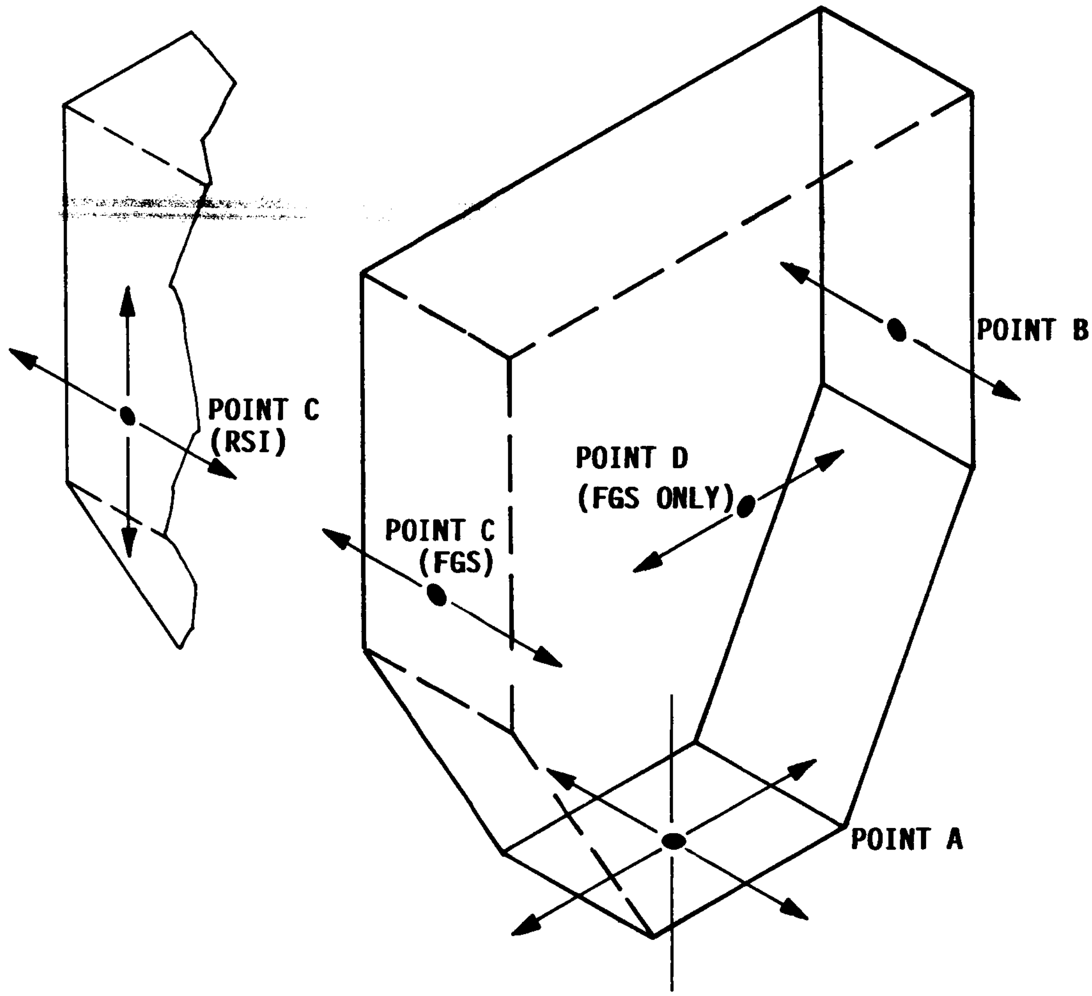
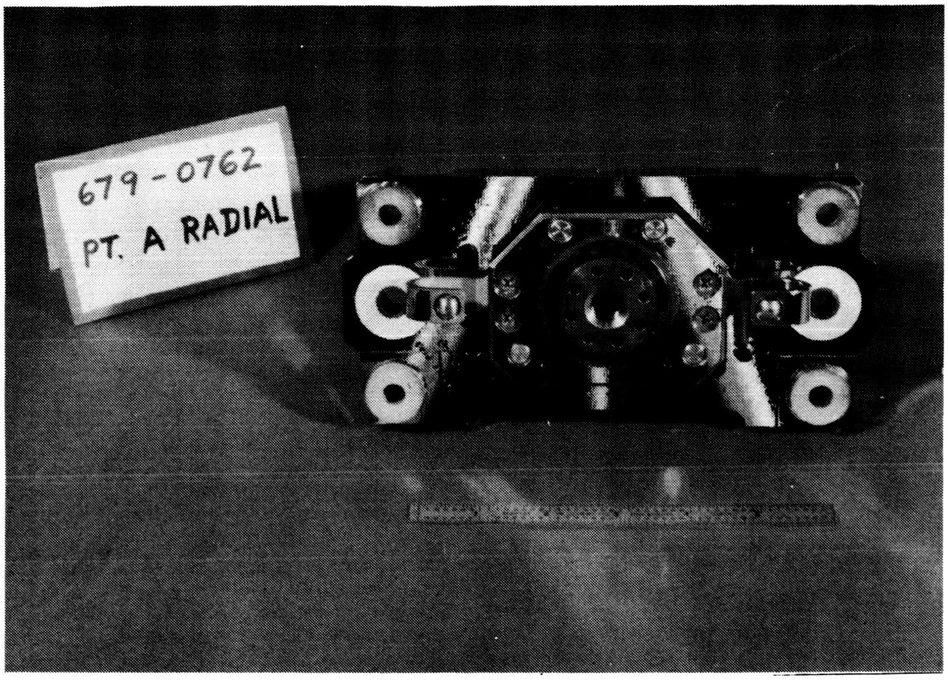


Figure 6. Radial Science Instrument (RSI) and Fine Guidance Sensor (FGS) registration fitting constraint directions



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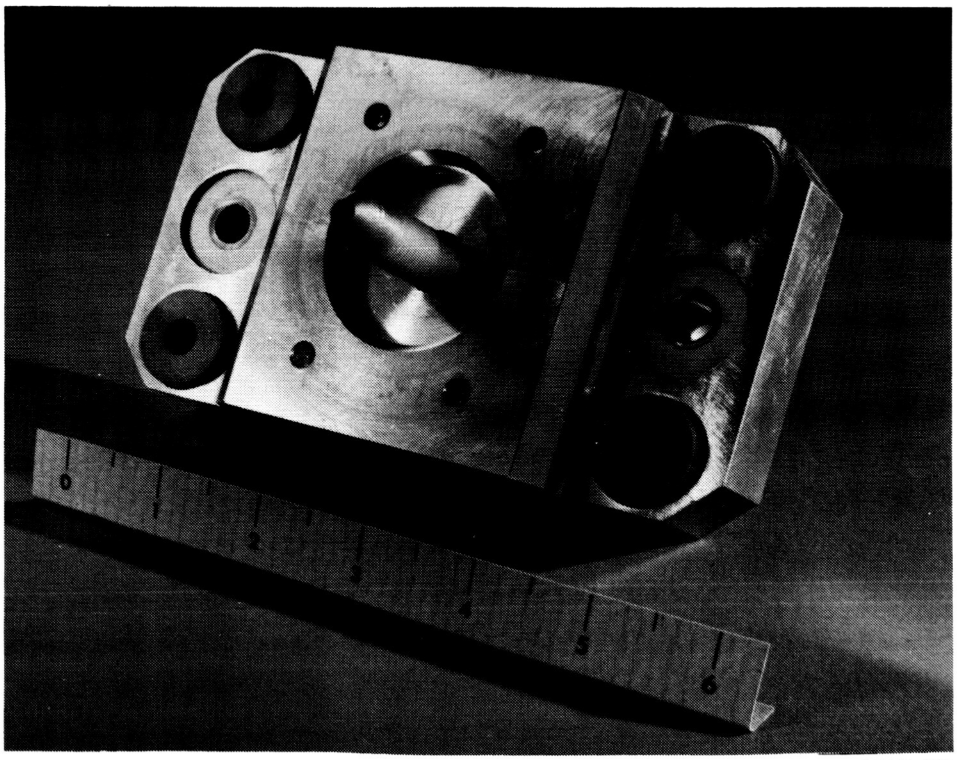

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Figure 7. Radial instrument "A" registration fitting pair

While design details vary, the Fine Guidance Sensor "B" and "C" fittings (Figure 8) are representative of design implementation.

The Radial Instrument "C" registration fitting differs from the "B" fitting in that it is a double constraint fitting incorporating two, orthogonal,flat and flexure pairs.

Only the "A" registration fitting requires "actuation" to remove or install a Radial Science Instrument or Fine Guidance Sensor; all other fittings self-align and engage as the instrument is radially translated. The "A" registration fitting pair screwdrive is driven through a solid drive rod extending through the instrument to its periphery where the crew member applies the required 60 NM (44 ft-lb) of actuation torque. As the pair is mated, the flexure springs are deflected resulting in the application of the registration preload of 1245 N (280 lb) across the "A" fitting pair.

REALIZED PERFORMANCE

Prior to delivery of the Optical Telescope Assembly in November 1984, development and verification tests were completed and confirmed the ability of the registration fittings to repeatedly position the EVA replaceable instruments within system requirements.

Realized worst-case despace, decenter, and tilt performance for each of the three instrument types were measured using instrument structural simulators having flight configured registration fittings. The data presented is the maximum-recorded repeatability error observed in three to five installation and removal sequences.

MAXIMUM-MEASURED REPEATABILITY ERROR

INSTRUMENT	DESPACE (μ m)	DECENTER (μ m)	TILT (arc sec)
Axial Scientific Instrument(S)	15.3 (0.0003")	15.3 (0.0013)	2
Radial Scientific Instrument	20 (0.0008")	38 (0.0015")	5
Fine Guidance Sensor(S)	36 (0.0014")	33 (0.0013")	4 Tangential Axis 4 Radial Axis

NOTE: Unless otherwise stated, decenter and tilt values are the maximum-measured values from one of the two separate applicable orthogonal components of error.

Crew systems compatibility has been verified in neutral buoyancy testing conducted at the Marshall Space Flight Center. The test program used high fidelity simulators of the telescope/instrument structures and registration fittings that were mechanically identical to flight hardware. One observation made during the neutral buoyancy test program was that since most of the structures and instruments are coated flat black for stray light control, it

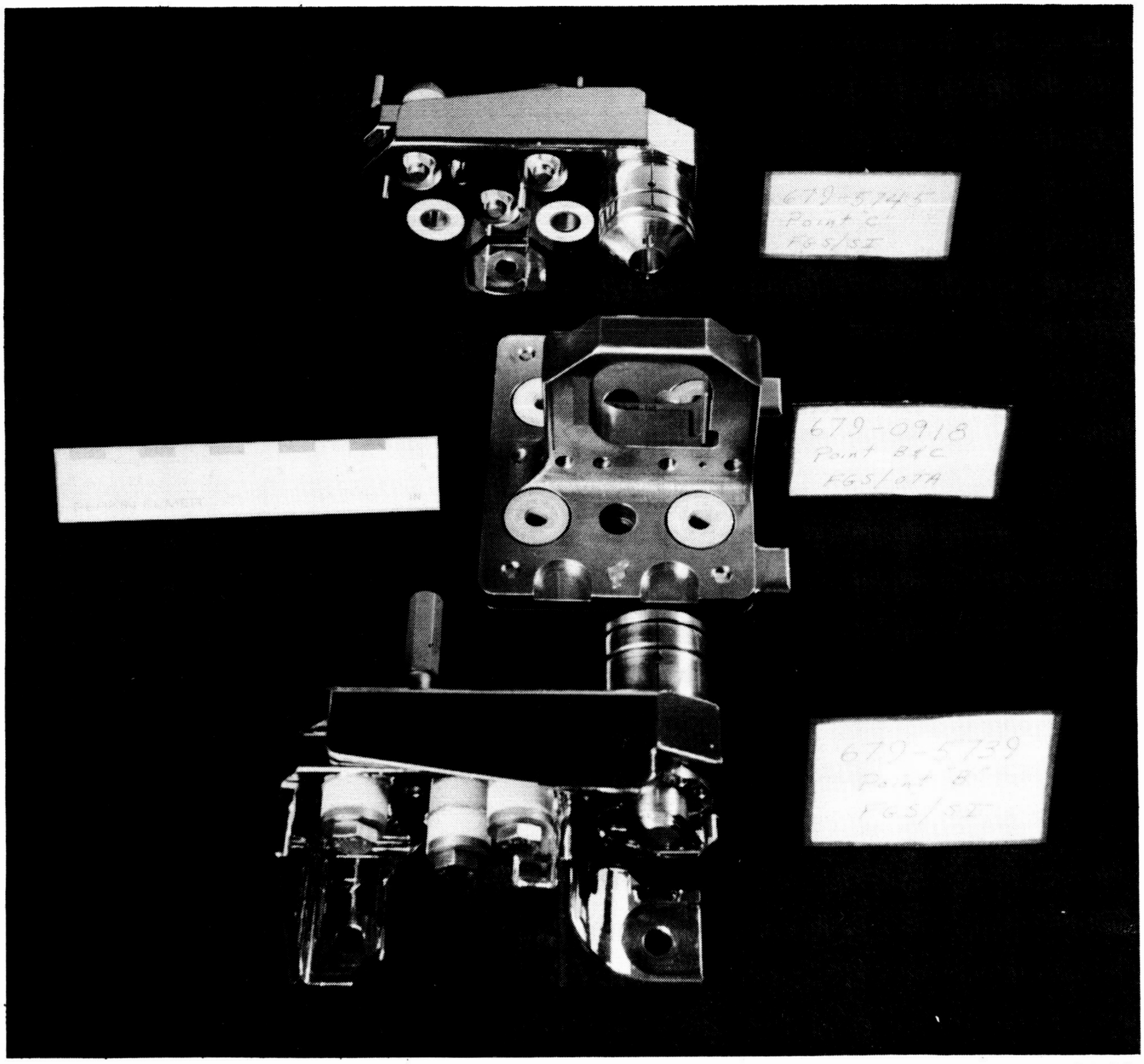

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Figure 8. Fine guidance sensor "B" and "C" fittings

is difficult for a crewmember to observe their position. A system of switches and verification lights was subsequently added to the flight hardware design as an aid in establishing proper positioning of the instruments and during the installation process.

CURRENT PROGRAM STATUS

The full complement of instruments has been installed in the telescope and the integrated vehicle is proceeding through functional and environmental testing.

PHYSICAL AND FUNCTIONAL PARTITIONING FOR IMPROVED SPACECRAFT SERVICEABILITY

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Interface standards are extensively employed by the aircraft and airline industry as building blocks for avionics architectures. Standards for interfaces are established for interface busses, power, instruction set architectures, heat dissipation envelopes, and physical interchangeability. These standards are developed and embraced jointly by the developer and operator. The ARINC Companies have been a primary participant in this achievement. We believe that the ARINC Companies airline experience provides the model for the space industry through the year 2000 and beyond.

Interface standards are required now for spacecraft. The Air Force has implemented policies to ensure that standardization of interfaces is included in on-space refueling and repair.[1] NASA awarded a contract to for the development of a fluid coupling connector [2] which promises to become a defacto standard. NASA has had successes in on-orbit repair and have recently completed several studies and workshops on this subject. The AESS Newsletter of September 1984 featured NASA's "Lessons Learned From Solar Maximum Repair".[3]

Significant progress in the development and application of standards will not be made until industry achieves consensus on the physical and functional partitioning of spacecraft subsystems. Benefits can then be gained from the establishment of interface standards which will extend to and beyond on-orbit servicing.

BACKGROUND

On-orbit servicing is of primary importance to DoD and NASA. Current satellite systems are having on-orbit maintenance accomplished. Future systems are being planned with on-orbit maintenance and servicing required. The Washington Post implied the SDI demands on orbit maintenance.[4]

These concepts will have an important impact on commercial applications of space systems. Demonstrations of on-orbit retrieval and repair have been accomplished. As the desired result, many are aware of the successes, also the problems encountered. Both Westar and Palapa were retrieved. The Solar Maximum Mission satellite's attitude control system and coronagraph polarimeter subsystems were replaced in the shuttle's cargo bay. Earlier, the Skylab crewmembers accomplished scheduled and unscheduled maintenance activity on-orbit. Maintenance has been performed during missions of the Satellite Transportation System. The Soviet Union's space program includes numerous examples of repairs and servicing, with a recent fluid transfer operation in the news.

The economic benefits of on-orbit servicing have been evaluated by analytical techniques including NASA cost and operational effectiveness models. DoD has a computerized spreadsheet capability (SATSERV) for examining economic and logistics aspects of launch and on-orbit servicing. Such models and techniques have been used to project the effectiveness of servicing space systems. H. O. Builteman concluded that the user community could avoid $13,000,000,000.00 in cost through 2005 through the use of on-orbit servicing.[5] This projection did not the Strategic Defense Initiative (SDI) concept.

Satellite servicing analysis requires definition of the categories of space systems. The physics of the problem differentiate between high and low earth, geosynchrous, polar, and orbits at various inclinations, etc. Satellite systems are stable, rotating, etc. As such, an analysis of these categories is required, for the current and the future timeframe. NASA, DoD, NOAA (National Oceanic and Atmospheric Administration), and U.S. commercial and foreign institutions recently projected 63 missions between 1986 and 1993 that can be reached by, or can fly down to, the Space Shuttle Orbiter.[6] Of these, 33 were deemed to be potentially serviceable on-orbit; 12 on a regular or scheduled basis. The Orbiter currently has an altitude limitation of 340 miles[7] and is limited during the powerless glides back to earth in cross range to less than 1,000 miles. Operational analysis has indicated that it will not be feasible to maintain a large operational system, such as SDI, without on-orbit servicing capability.

Space system design is key to the accomplishment of such servicing and repair actions as stated by H. T. Fisher, "... the dissemination of succinct, easily understood, and well illustrated design guidelines to assist the total systems and design team in the development and evolution of an easily serviceable system." [8]

The specific intent of this paper is to propose that a Spacecraft Interface Standard is a specific and important component of the design guidelines as described by Fisher.

This will lead to a degree of standardization in design while still providing the flexibility required to accommodate the state-of-the-art technology; will enhance insertion of new technology into on-orbit systems through modular subsystem replacement; and will facilitate on-orbit servicing.

FORM, FIT, AND FUNCTION (F^3)

In the aircraft business, we started the standardization program with form and fit requirements, and developed an information transfer bus that allowed electronic devices to interface and communicate. It was found that it is difficult to size a standardized replaceable unit and its communications bus until its function has been defined. In the case of space systems, the replaceable unit will be called an Orbital Replaceable Unit (ORU). The problem was solved for aircraft by defining the smallest possible functional building block, then allowing for modular increases in size. The size limit of the largest Line Replaceable Unit (LRU) was based on human factors considerations. In space greater masses can be considered.

The objective is to establish spacecraft systems form, fit, and function (F^3) standards. A function such as a power supply, mass memory unit, or fluid transfer pump would meet volumetric, heat dissipation, attachment, and electrical interface requirements. The designer can choose details and concepts of the ORU design, however the part will be interchangeable. Design flexibility to accomodate alternative concepts and technology is encouraged.

Full F^3 standardization has not been achieved in the aircraft business. Commercial airlines have had the most success with F^3 standards established for roughly 70% of the avionics suite. The military has less F^3 standards experience, however form, fit, environmental, and bus interface standards are being pursued.

Avionics Interface Design Standards have been developed for the U.S. Air Force by the ARINC Companies. Proposed DOD-STD-1788 provides for F^3 standards for aircraft. Figure 1 illustrates the maintainability benefits of such standardization, using the F-15 aircraft avionics bay as the example. The figure shows the current avionics bay configuration. Also shown is how the installation would look if DOD-STD-1788 had been imposed.[9] The standard requires rear-mounted, low insertion force connectors and clutch-equipped extractor/hold-downs for ease of box replacement.

Reliability factors are also addressed in DOD-STD-1788. Thermal environment, vibration, load, other reliability related interfaces between the aircraft and the avionics are defined. The new standared is currently being used on several military aircraft which are now in the design stage.

The airline precedent to DOD-STD-1788, ARINC 600 [10] is used on the Boeing 757 and 767, the Airbus 300 series, and the Douglas Super-80 aircraft series. The definition of interfaces contained in aircraft-related interface standards, such as MIL-STD-1553[11] and ARINC 429 [12] needs to be established for spacecraft systems.

Progress is being made. Figure 2 is from the NASA sponsored Satellite Service Handbook - Interface Guidelines [13] A marked similarity exists between the ORU, the aircraft avionics rack, and the DOD-STD-1788 panel-mounted LRU. NASA has established installation design parametrics such as human factors clearance envelopes and hold-down devices.

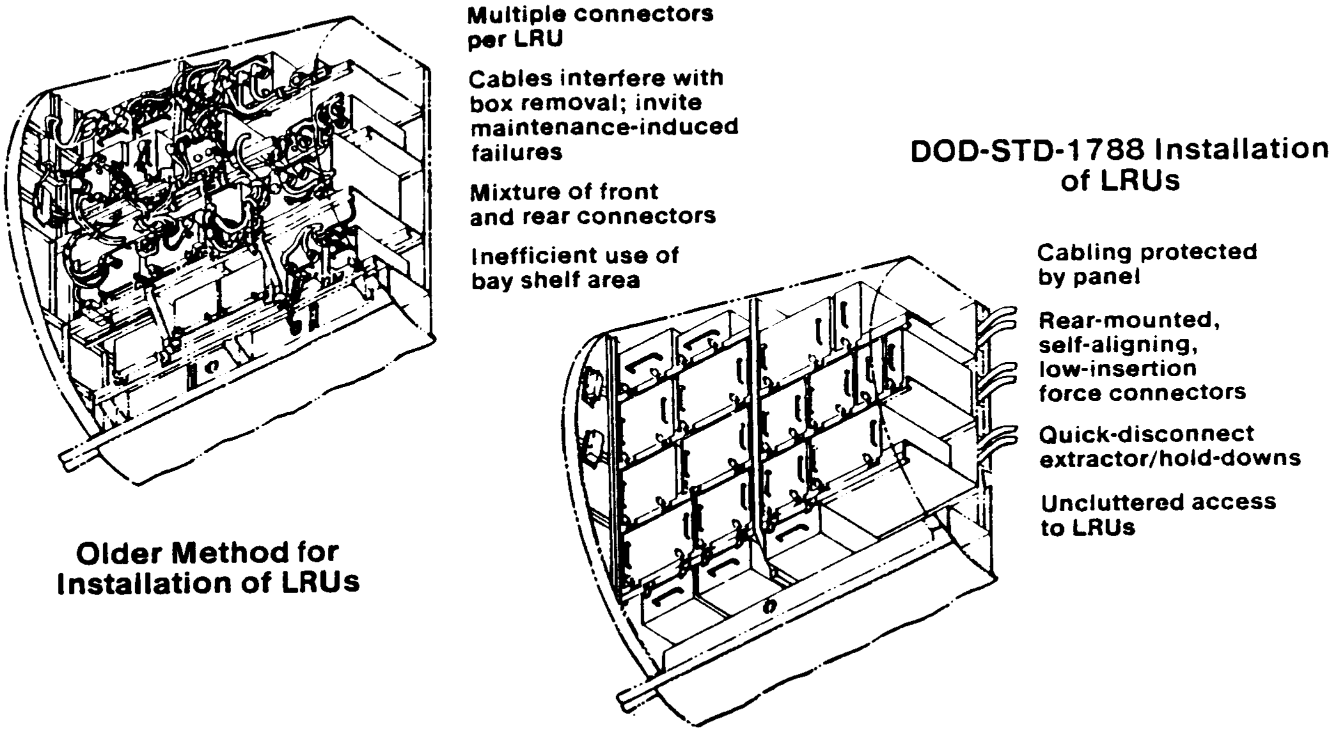

Figure 1. Packaging and Maintainability Benefits of Interface Standardization

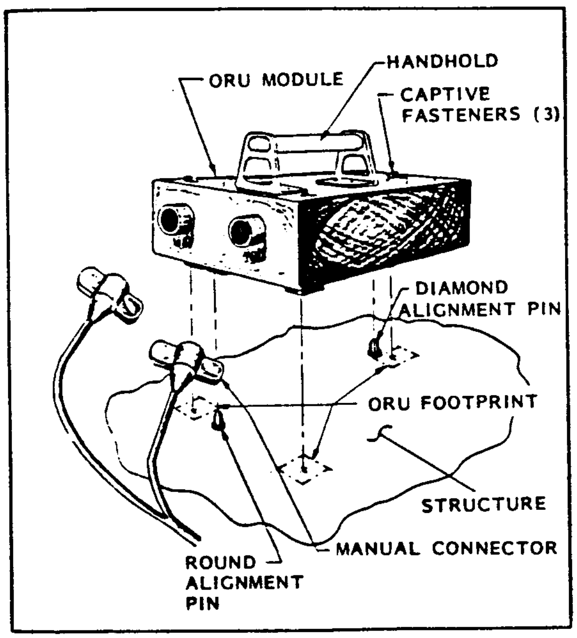

Figure 2. Orbital Replaceable Unit (ORU)

The primary interface standards are for:

- Mechanical interfaces (indludes connectors)
- Thermal envelopes/interfaces
- Power interface
- Information transfer bus
- Functional partitioning
- Testability interface
- Service access/human factors considerations
- Electrostatic discharge protection

Functional partitioning and testability are related and critical to the success of on-orbit servicing. The aerospace community tests by function. When functions are distributed over more than one ORU, unambigious fault isolation is difficult. The determination of the ORU functions and testability are critical technical issues. Analytical tools are available to examine the testability consequences of alternate partitioning strategies.[14] The technology of defining built-in-test versus remote maintenance monitoring (RMM) requirements is well developed.[15]

ADHERE TO THE STANDARD

The willingness of the designer to adhere to the defined standards must be recognized in the development of the standard. The lesson learned by the ARINC Companies is that the process for establishing the standard must include the designers of the hardware[16]. The first precept is that the standard must be established in an open-forum environment including the users and suppliers. A neutral arbitrator (not a user or supplier) must be selected to recommend a position when consensus cannot be achieved. The ARINC Companies unique non- user and non-supplier position and the reputation of being unbiased and impartioal has allowed effective utilization in the role of the neutral participant during the development of a large number of commercial and military standards.

We have found consensus in industry partitioning and packagaging methodologies. Differences are ofter just the result of arbitrary decisions. One firm may use a 4" x 5", 250 pin electronic assembly, while another uses a 3-1/2' x 6", 225 pin unit. The differences are just large enough to cause the operator and maintainer to pay separately from limited resources for the development, operation and logistics of each.

Consensus can be achieved. The suppliers and operators both must perceive potential benefits from the standards. In the aircraft industry, suppliers have found that standards permits participation in a larger marketplace. Interface, rather than piece part or detailed design standards permits the designer to be innovative to achieve a competitive advantage. Operators will have access to more than one supplier which provides the choice of more than one product. Cost and reliability benefits which has resulted from interface standards in the military and commercial avionics communities have been extensively documented.

BENEFITS OF INTERFACE STANDARDS

Increased on-orbit servicing, improved reliability, improved system availability; reduced system development, acquisition, operational, and maintenance cost; new technology insertion capability, system lifetime extension, system survivability enhancement, and common maintenance support services are potential gains to the space system user obtainable from interface standards application.

Increased Capability For On-Orbit Servicing

Increased standard modules, physical and electrical interfaces will enhance the design of on-orbit servicing systems, tools, equipment, procedures, and human interfaces. Those servicing aids will be useful for an expanded set of space systems and will be applicable to DoD, NASA, and the commercial world. The increased use of on-orbit support services, aided by standardized modules into space system design, will have the effect of reducing the non-recurring cost allocated to any specific customer or on-orbit support action. Increased affordability will increase the incentive to design systms that can benefit from the maintenance support services.

Improved Reliability and Reduced Production Cost

Commercial airline and electronic equipment production esperience shows that operational reliability improves as production quantity increases. Labor intensive manufacturing processes give way to automation. The design is improved to eliminate problem areas. The same result is expected for standardized space system modules. Improved reliability of several orders of magnitude is foreseeable. The production of F^3 specification modules by more than one vendor provides the choice of designs offering reliability or other beneficial features.

Reduced production cost is expected for a standardized module which is reflective of the amortization of the non-recurring design, production setup, and other costs over a higher production base; and the influence of competition. A price reduction of one-third has been seen as the result of the application of F^3 standards in the electronics community.

Improved System Availability and Space System Survivability

F^3 modules serve to enhance system availability. The capability to reconfigure or switch functions between several modules through the use of a standardized data bus architecture, as a result of commands or autonomously, will increase with standardized modules. Automonous reconfiguration through the use of on-board RMM information can contributor to system survivability and battle damage repair.

Reduced System Development, Acquisition, Operational and Maintenance Cost

An inventory of off-the-shelf F^3 specification modules will serve to reduce the expenditure of resources and time required to develop a space system. Increased quantities of reduced price and increased quality F^3 modules will increase the selection for use by the designer and supplier. The development cost is largely eliminated, and the acquisition cost is reduced when the off-the-shelf standard module is ordered. Operation and maintenance costs is less for the standardized than unique equipment due to factors including the existence of test and repair services. If spares are required on-orbit, the quantities of unique items will be reduced.

New Technology Insertion Capability

Standardized interface specifications, defined physical envelopes, and a standard data bus will enhance the development of improved performance modules with reduced power consumption, increased reliability, etc. These may be used to replace outdated F^3 modules during on-orbit servicing.

The use of the improved modules can also assist in the extension of the useful operational lifetime of a satellite system. The amortized annual cost would be reduced as the life is extended over more years of useful service. Standard modules also enhance the concept of on-orbit assembly of space systems and structures.

Common Maintenance Support Services

The space system maintenance support services can be combined with those supporting other space systems, whether these services are at ground or on-orbit. This will lead to cost and quality improvements in these functions.

SUMMARY

Affordability is the critical issue. Military and civil space system users count the cost of placing a new capability in space. Previously, the transportation (launch) cost and capability was of primary importance. Reduced launch costs now make it possible to consider the next approach to reducing the cost and increasing the effectiveness of the satellite system. To the extent that standardization will extend the useful life of the spacecraft through improvements in reliability, reduncancy, seviceability, and insertion of improved technology, the development and transportation costs can be amortized over more years of operation and more space system applications. A greater degree of modularity will permit incremental launch concepts for larger systems.

Increased standardization is possible in a high technology environment. The form of standardization is the critical issue. Draw on the experience and success achieved in the avionics comunity to help us in getting this worthwhile endeavor underway.

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LEASECRAFT SYSTEM

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The LEASECRAFT concept provides the space and ground infrastructure for space commercialization. Figure 1 depicts the sequence of payload exchange which only requires that the payload and LEASECRAFT docking system be launched for repair or payload changeout missions. The platform remains in orbit and is maintained during routine visits. The functional requirements which were established early in the LEASECRAFT program are summarized in Table 1 and resulted in a system design that is shown in Figure 2. Of the six major system elements, two are controlled by NASA and one by the customer. One of the LEASECRAFT challenges is the integration and coordination of these diverse elements to meet the customer requirements. The "bullets" of Figure 2 highlight the major components of the specific element.

The space platform consists of three Multimission Modular Spacecraft (MMS) modules, Attitude Control, Communication and Data Handling, and Power, which are designed to be replaced in space. The platform also includes a hi—gain tracking antenna, and solar array assemblies which are designed for on—orbit maintenance. All of the preceding components and the propulsion subsystem are attached to a structure which interfaces directly with the Shuttle and are interconnected by an internal harness. Figure 3 shows the multiplex data bus and power bus configuration. This MMS derived system provides a very simple interface for the payload and platform modules. It is currently in use on the Solar Maximum Mission, Landsat 4 and 5, and the Gamma Ray Observatory (GRO) Program.

Table 2 summarizes the performance of the LEASECRAFT space segment. As indicated the attitude control subsystem (ACS) is very capable and with the use of a payload sensor can provide exceptional performance. The modular power subsystem (MPS) provides unregulated power and can be expanded by the addition of more power modules and solar array. The propulsion module provides the mobility for platform orbit adjustments and Shuttle rendezvous. The communications and data handling module (C&DH) contains the brain of the spacecraft and communicates with the LEASECRAFT control center. The payload weight is constrained by the Shuttle lift and down—weight capability, the altitude changes desired, and the slew rates desired. Payloads as large as 60,000 pounds can be accommodated. The platform lifetime is not limited by any consumables, since LEASECRAFT can be refueled. With maintenance LEASECRAFT will be operable as long as economically feasible. Figure 4 shows the LEASECRAFT configuration which was proposed to NASA for their platform services contract. Three payload interfaces are available and payload power levels up to 2000 watts can be provided. The "Z" payload location provides more than 88 square feet of unconstrained area and can be further expanded for specific missions. The "X1" location provides more than 32 square feet of interface area and has a depth which is determined by the launch mode (attached or not) and the center—of—gravity constraints. The "X2" payload location is an MMS type module and has a 16 square foot interface and is nominally 1.5 feet deep. The X2 module can only be replaced at the present time by an EVA operation while the others can be replaced with the Shuttle arm.

Figure 5 depicts a possible configuration of a materials processing payload with a NASA payload in the X1 location. This configuration has a larger solar array and an additional power module. Figure 6 provides more detail on the MMS module box accommodations.

The LEASECRAFT platform is designed to be launched by the NASA Space Transportation System. The initial launch and first revisit will be provided by NASA under the terms of the NASA/Fairchild Joint Endeavor Agreement which provides a free launch for the LEASECRAFT platform and its commercial payload with a service flight six months later. This service flight will demonstrate rendezvous, berthing, payload changeout, module changeout and other appropriate tests. Key to the success of commercial operations will be routine and reliable schedules of STS service at cost that can be established early enough to provide financial certainty. Since transportation cost will be the largest recurring cost to commercial users, the Shuttle charges are determined by length, weight, requirements for non-standard services, and the deployment and rendezvous altitudes required by the payload. LEASECRAFT has orbit adjust capabilities, so a cost/risk tradeoff can be made relative to the STS service altitudes.

The payload changeout and module service approach both utilize a LEASECRAFT docking system to attach LEASECRAFT to the Orbiter so that the LEASECRAFT remains outside of the cargo bay. The payload on the LEASECRAFT is then relocated to an interim berthing location and the new payload attached to the LEASECRAFT. The old payload is then attached to the payload carrier which carried up the replacement payload. Module servicing is conducted in the same manner with a module that is carried up on the docking system.

Table 3 itemizes the standard and optional services that are provided by the LEASECRAFT System. A very attractive feature for commercial users is the payment-for--services-rendered philosophy. The Fairchild Space Operations Company finances the development of the LEASECRAFT System which allows the user to limit his cash flow by not having to make an investment in a platform. The platform cost is then treated as an operational expense. The deferment of these costs can solve budgeting problems for all users. NASA initiated a procurement in January which would require the services of a privately owned and operated multipurpose space platform. They asked for 60 months of service over an 84 month period to accommodate three or four "Explorer" payloads. Service is to begin in late 1988. Fairchild submitted a proposal to NASA which provided excess capability. This excess capability takes two forms. First, the twenty-four months that NASA is not using the facility and second, platform capabilities in excess of NASA's requirements. In the first case all of the platform capabilities described in Table 2 are available and in the second from 100 to 1000 watts of power are available during the NASA missions. Payloads flown in conjunction with the NASA missions will be constrained by the NASA mission operation requirements; however, significant payload opportunities do exist during these periods at reduced cost.

Figure 7 depicts a docking system that permits the LEASECRAFT to attach to the Shuttle with only minor requirements for cargo bay space. This structure also provides an interim location for payload storage thereby solving the "third hand problem." The alternatives to this type of system are the use of a second RMS which would mean committing 2/3 of the RMS inventory to one Orbiter and the necessary installation time, or reserving sufficient cargo bay space to install LEASECRAFT in the bay for payload changeout and servicing. Both of these alternative approaches greatly complicate manifesting and cost more than the "docking mast" approach. These considerations are of great importance to the commercial STS customer.

The STS transportation cost and platform design are also driven by the rendezvous altitude and servicing scenario. The STS "standard scenario" requires the platform to be at 260 n.m. (a one day repeat orbit) at the time of STS launch and then to descend to 170 n.m. for rendezvous. The platform then requires a substantial propulsion system and refueling capability. This capability duplicates STS capabilities. Alternative scenarios can minimize platform propulsion requirements by having the STS fly to 260 n.m. but cost about 10800 lbs. of STS performance (relative to 160 n.m.) and will approximate the requirements of the Space Station. Other options will exist for specific mission requirements. These scenarios are shown in Figure 8.

In summary, LEASECRAFT offers many advantages to the user and will promote space commercialization. These advantages include:

END–TO–END COMMERCIAL SERVICE

PAY FOR SERVICE RECEIVED

ON–ORBIT REPAIR

CHANGE–OUT OF INSTRUMENTS

LARGE PAYLOAD CAPACITY

HIGH HERITAGE HARDWARE

PLATFORM MOBILITY

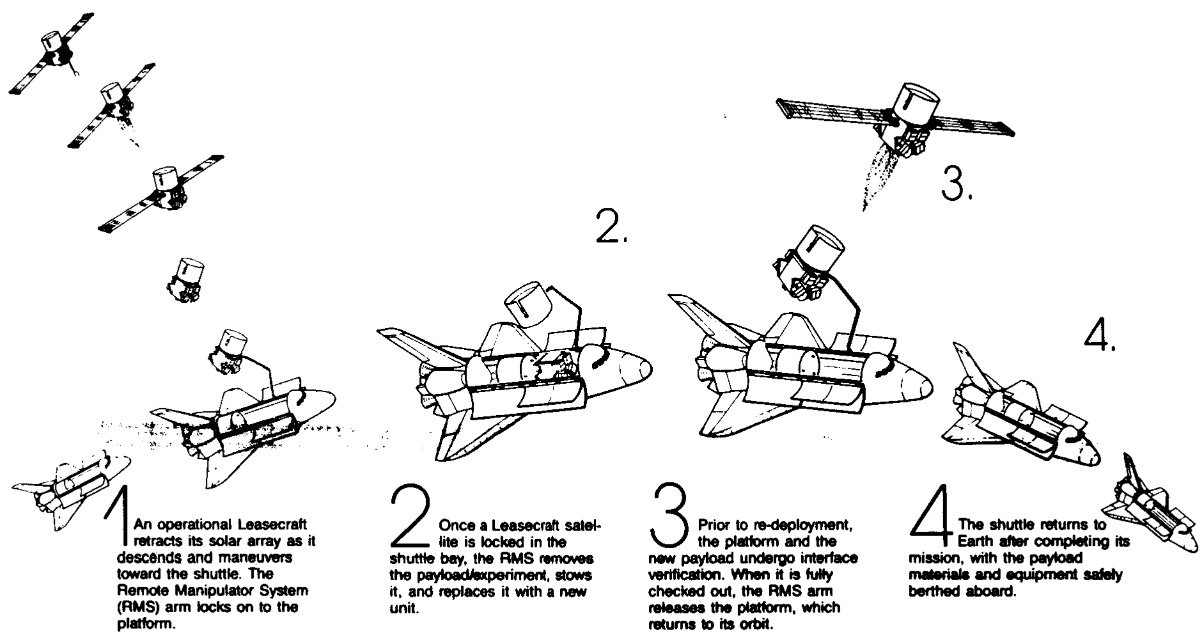


Figure 1. In-Orbit Operational Sequence

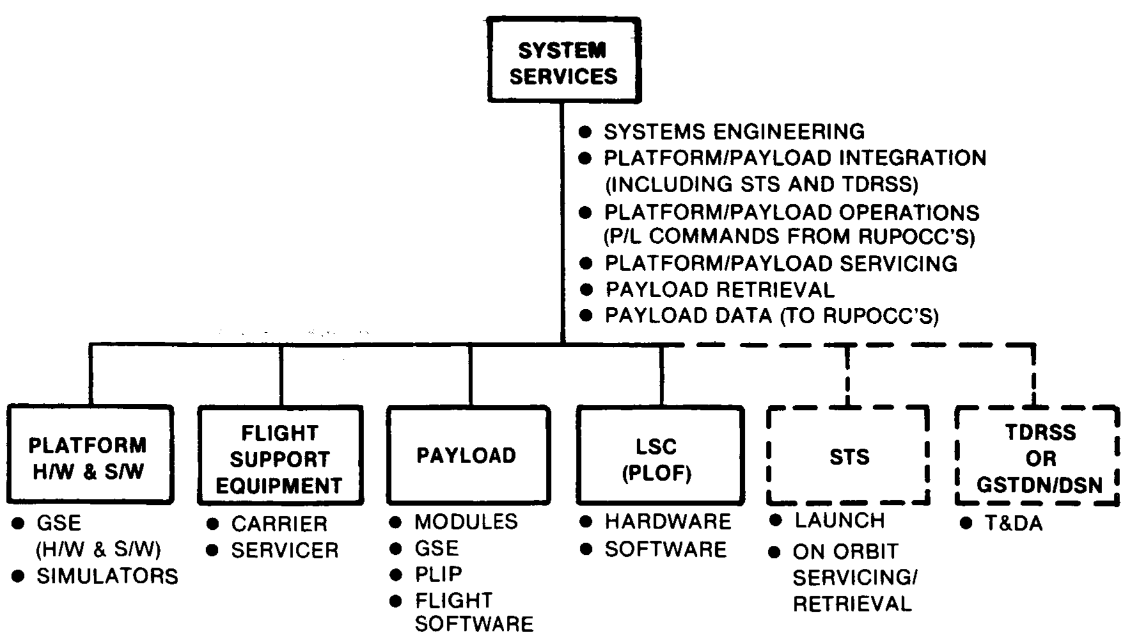


Figure 2. Leasecraft Platform Services System

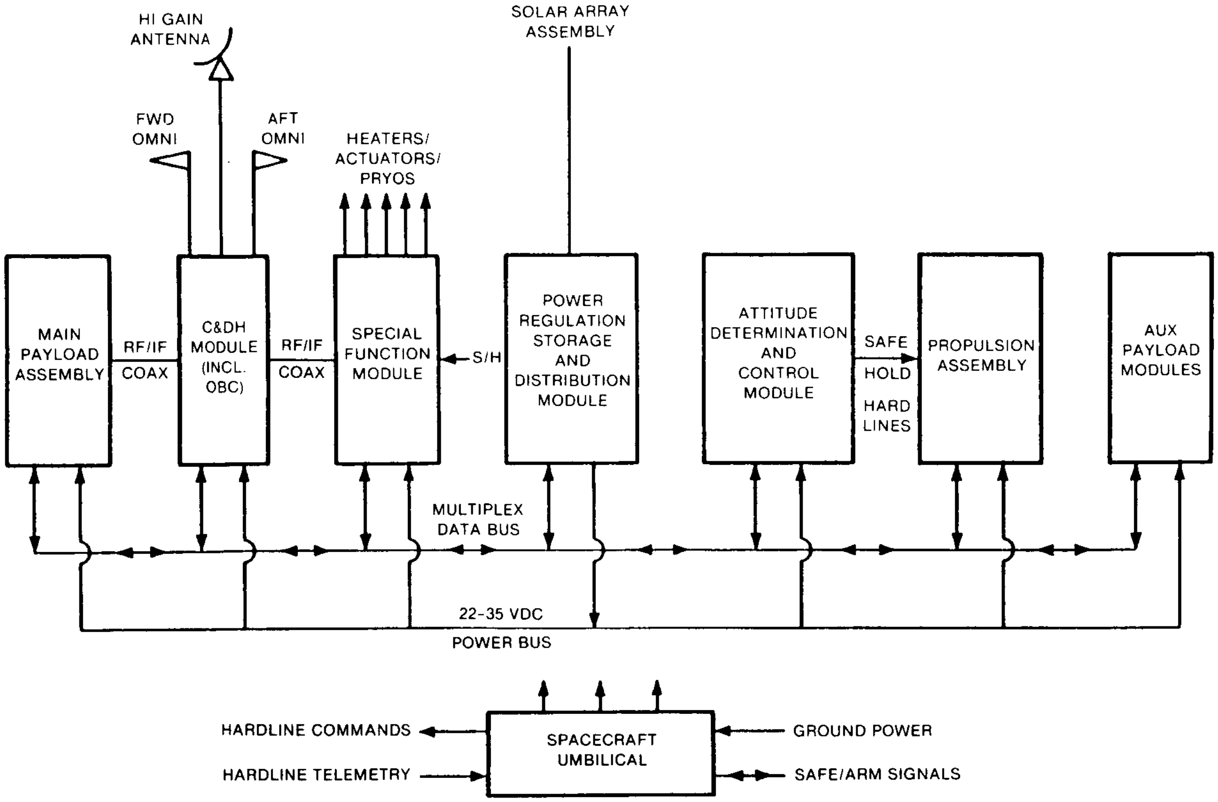
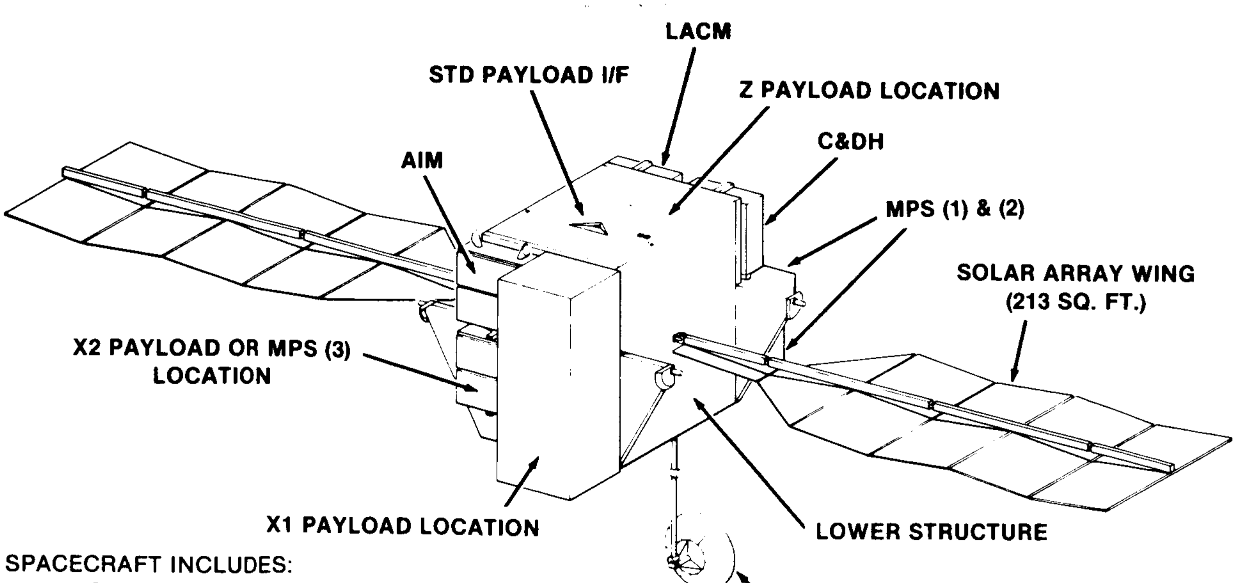


Figure 3. Leasecraft Electrical Configuration



SPACECRAFT INCLUDES:
(1) C&DH
(1) LACM/AIM
(2) MPS
(1) PROPULSION MODULE
(2) 213 SQ. FT. SOLAR ARRAY WINGS
(1) HIGH GAIN ANTENNA
(1) SPECIAL FUNCTION SUBSYSTEM

Figure 4. NASA Leasecraft Configuration

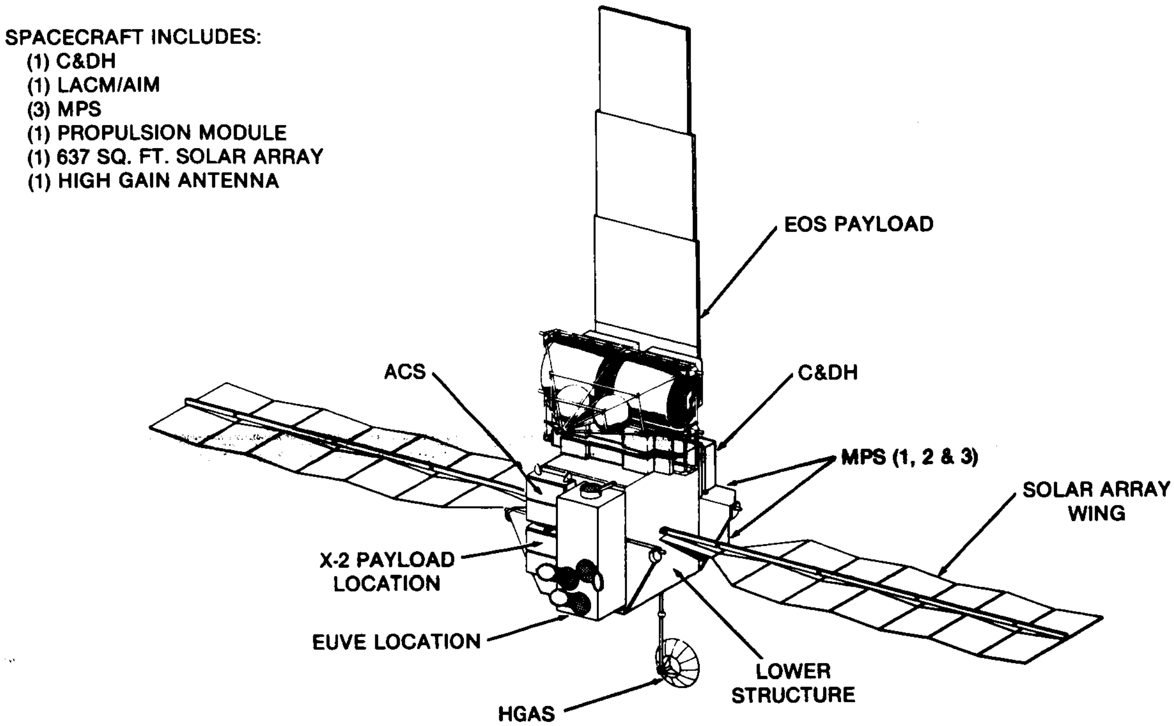


Figure 5. Materials Processing Configuration

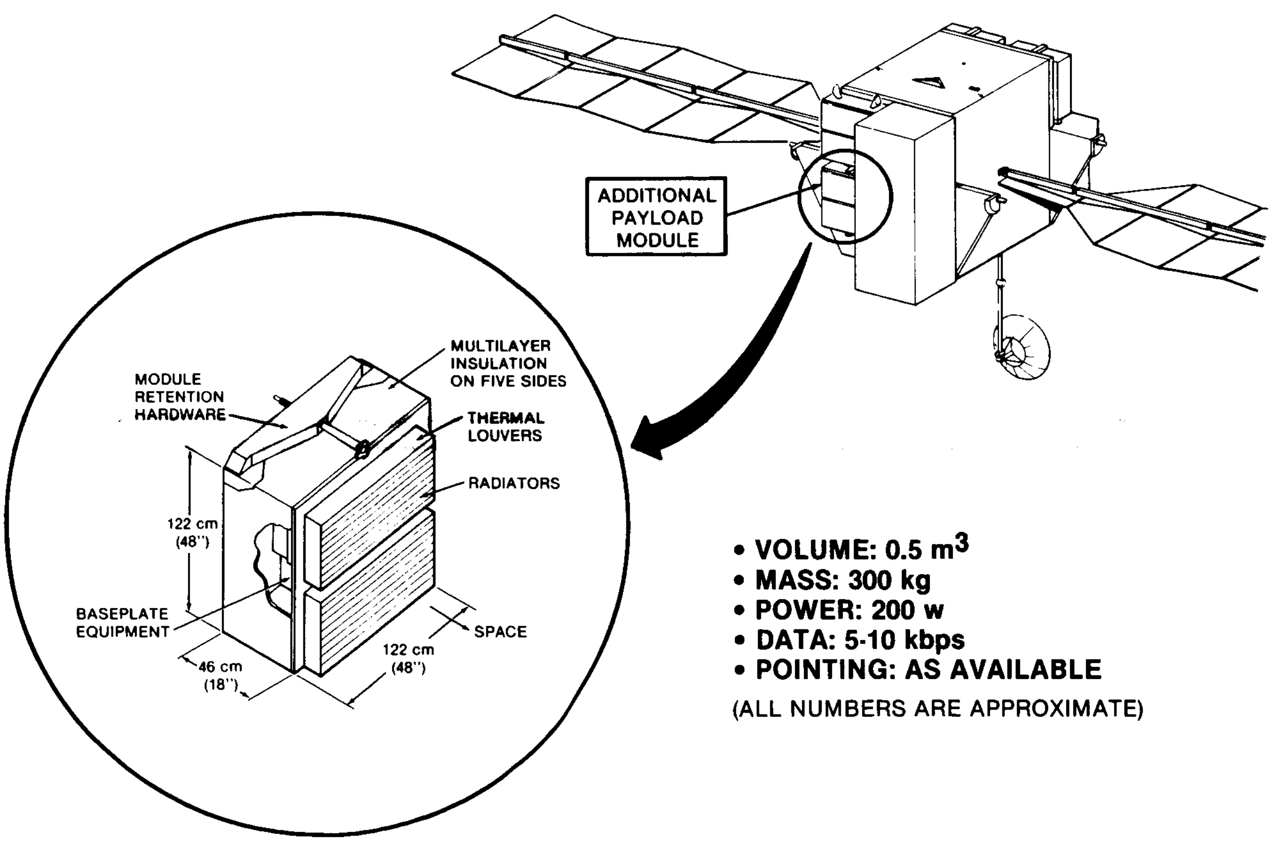
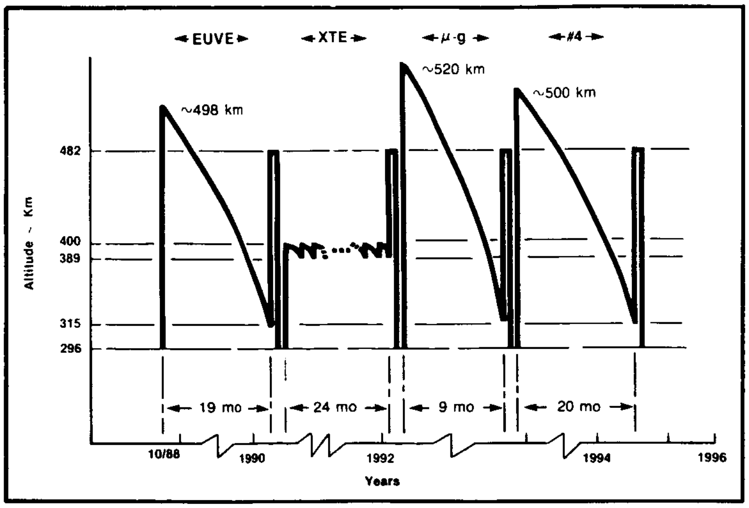
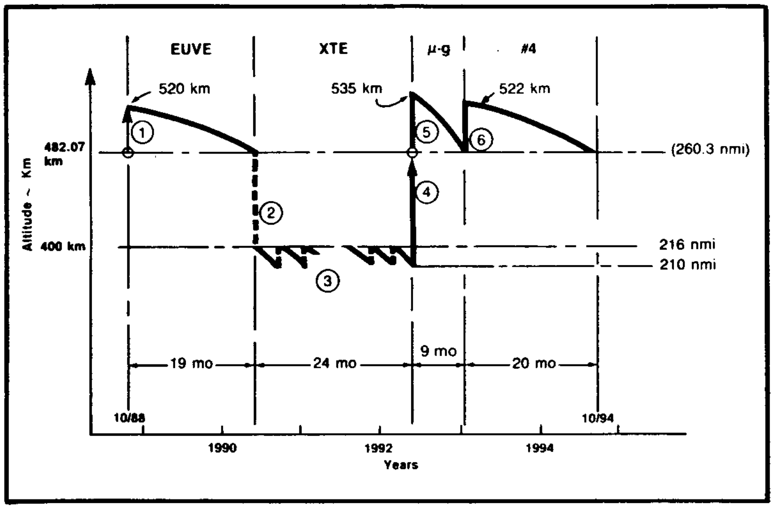


Figure 6. Module Box Accommodations

NASA STANDARD RETRIEVAL OPERATIONS (SCENARIO 1)



260.3 NMI OPERATIONS (SCENARIO 2)



216 NMI INTERMEDIATE ORBIT OPERATIONS (SCENARIO 3)

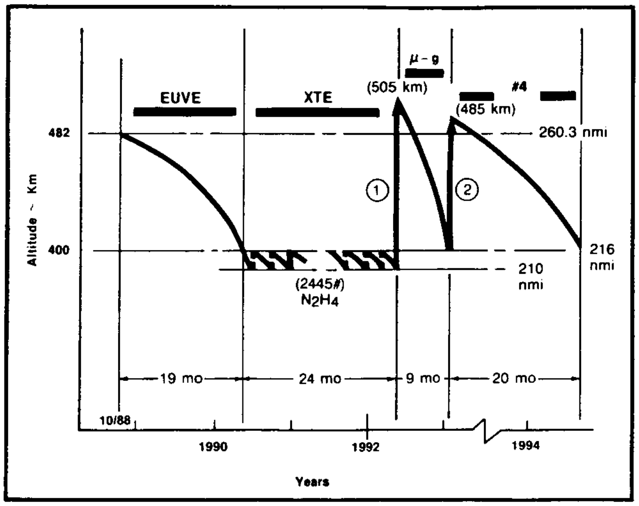
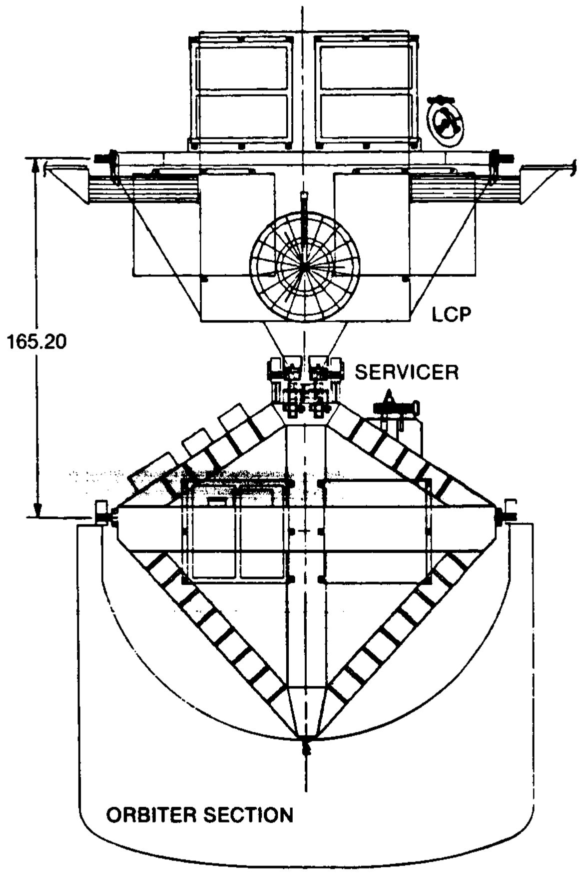


Figure 7. The Leasecraft Docking System



ORBITAL REPLACEABLE UNITS
PAYLOAD(S)
C&DH MODULE
POWER MODULES
ACS MODULE
THRUST CHAMBER - 100 LBF
HIGH GAIN ANTENNA

TBD:
SOLAR ARRAY
SOLAR ARRAY DRIVE
SPECIAL FUNCTION UNIT

Figure 8. NASA Platform Services Rendezvous Options

Table 1. Leasecraft Functional Requirements

- Up to 6,600 watts of electrical power in 1650W increments
- Two-way transfer 20,000 lbs. to 360 n. miles; 4,000 lbs. to 600 n. miles
- TDRSS, STDN or SGLS compatible communication links
- **Attitude error <.01° with attitude rate <.002°/sec**
- Autonomous operation
- Dual redundant (no single point failures)
- Capable of being launched & retrieved by STS
- Cost-Effective utilization of STS cargo bay (1,100 lb./ft.)
- Direct spaceframe attachment to STS longeron & keel fittings
- Utilize standard MMS modules
- All modules & major sub-assemblies exchangeable in space environment
- Leasecraft mated to payload in orbit
- Mating, deployment, retrieval & changeout timeline's minimized
- Maximum use of RMS for service operations

Table 2. Leasecraft Performance Summary

PAYLOAD WEIGHT CAPABILITY	PRIMARY PAYLOAD: UP TO 14,500 kg (32,000 LBS.) SECONDARY PAYLOADS: UP TO 1,000 kg (2,200 LBS.) } APPROXIMATE
TYPES OF MISSIONS	STELLAR, SOLAR, EARTH POINTED, OR SPECIAL PURPOSE MISSIONS; LOW EARTH ORBITS; INERTIAL POINTED OR PAYLOAD POINTED
OPERATING ORBITAL ALTITUDE	LOW EARTH ORBITS, ALL INCLINATIONS ≥28.5 DEG.
LIFE EXPECTANCY/REDUNDANCY	ALL CRITICAL ELEMENTS REDUNDANT; ALL SUBSYSTEMS REPLACEABLE IN ORBIT. NO SINGLE POINT FAILURE TO PREVENT RESUPPLY OR RETRIEVAL BY SHUTTLE.
LAUNCH VEHICLE	SPACE SHUTTLE FOR LAUNCH, SERVICE, AND RETRIEVAL.
COMMUNICATIONS AND DATA HANDLING SUBSYSTEM	
TRANSPONDER	S-BAND STDN/TDRSS, TRANSPONDER OUTPUT POWER AT MODULE INTERFACE 0.8, 2.0, 4.0 WATTS, SELECTABLE AT MANUFACTURE.
COMMAND RATES	2 KBPS (SHUTTLE/STDN). 125 and 1 KBPS SELECTABLE (TDRSS).
REAL-TIME TELEMETRY RATES	1, 2, 4, 8, 16, 32, 64 KBPS
TELEMETRY FORMATS	2 SELECTABLE PRIOR TO LAUNCH. PLUS IN-ORBIT PROGRAMMABLE CAPABILITY: ALL FORMATS CONTAIN 890 8-BIT DATA WORDS MAXIMUM
STORED DATA DUMP/MISSION DATA SOURCE	2.048 MBPS MAXIMUM. 1.024 MBPS CODED DATA. UP TO 100 MBPS IN OPTIONAL WIDEBAND DATA MODULE.
ON-BOARD COMPUTER	18 BITS PER WORD. 32K WORDS OF MEMORY, EXPANDABLE TO 64K WORDS. 5 MICROSECOND ADD TIME.
DATA STORAGE	10^8 BIT TAPE RECORDERS.
ATTITUDE CONTROL SUBSYSTEM	
TYPE	3-AXIS STABILIZED, ZERO MOMENTUM
ATTITUDE REFERENCE (WITHOUT PAYLOAD SENSOR)	STELLAR (INERTIAL)
POINTING ERROR (ONE SIGMA)	
WITHOUT PAYLOAD SENSOR	$<10^{-2}$ DEG.
WITH PAYLOAD SENSOR (IDEAL)	$<10^{-5}$ DEG.
POINTING STABILITY (ONE SIGMA)	
AVERAGE RATE	$<10^{-6}$ DEG./SEC.
JITTER	
WITHOUT PAYLOAD SENSOR	$<6 \times 10^{-4}$ DEG. (20 MINUTE PERIOD)
WITH PAYLOAD SENSOR (IDEAL)	$<10^{-6}$ DEG.
SLEW RATE	MAXIMUM 1.6°/SEC WITH STANDARD INERTIAL REFERENCE UNIT
POWER SUBSYSTEM (BASELINE - 1 MODULE - UP TO 5 MODULES AVAILABLE)	
VOLTAGE OUTPUT	$+28 \pm 7$ VDC
POWER TO PAYLOADS (MAX.)	1,000, 2,600, 4,200, 5,700, 7,300 WATTS (1 - TO - 5 POWER MODULES)
BATTERIES	TWO 20 - AMPERE - HOUR BATTERIES TO THREE 60 - AMPERE - HOUR BATTERIES PER POWER MODULE
PROPULSION MODULE	4 - TANK HYDRAZINE SYSTEM CAPABLE OF CARRYING 1800 Kg (4000 LBS.) 4 - 445N (100 LB.) ORBIT ADJUST THRUSTERS, 12-22.2N (5 LB.) RCS THRUSTERS.

Table 3. Leasecraft Services

Standard Services

Standard Services under fixed priced contract will include:

- Payload integration, launch, on-orbit operation and return
- Experimenter requirements accommodation analysis
- Interface documentation & basic engineering support
- Selected baseline flight hardware for integration with payload:
  - Remote Command and Telemetry Unit(s)
  - Standard electrical/mechanical interface elements
- Master interface tool for flight adapter for payload module
- Leasecraft/payload operations plan and flight software
- Platform, mechanical, power, and data system simulator(s)

Optional Services

Optional services under Mission Unique Contract can include:

- Additional platform power
- Attitude control augmentations
- Communications augmentation for higher data rates
- Additional systems engineering support and services for payload modules and instruments
- Shipping containers and transportation for payload modules and instruments
- Software for automated checkout equipment for payload modules
- Design, fabrication, integration and testing of payload modules
- Remote Data Work Staions
  - Supporting flight operations
  - Capture, display, and return to user of scientific data
  - Communication links to RUPOCC

ORBITAL MANEUVERING VEHICLE (OMV)

SATELLITE SERVICING

ORBITAL MANEUVERING VEHICLE (OMV)

The OMV provides for the extension of payload services and capabilities out of the Shuttle and the Space Station. These services include spacecraft delivery and retrieval to and from higher orbits, reboost or deboost, payload viewing and subsatellite support. The OMV will also be capable of supporting advanced mission kits for remote servicing, refueling, and debris retrieval.

ORBITAL MANEUVERING VEHICLE

- THE ORBITAL MANEUVERING VEHICLE PROVIDES EXTENDED PAYLOAD SERVICES OPERATING OUT OF THE SHUTTLE AND SPACE STATION
  - EXTENDS STS OPERATIONAL CAPABILITY
    - AUGMENTS PERFORMANCE IN LOW EARTH ORBIT
    - EXPANDS ENVELOPE OF MAN'S INVOLVEMENT IN MAINTENANCE/REPAIR
      - RETURN OF SPACECRAFT TO THE SHUTTLE FOR EVA OPERATIONS
    - NEW AND UNIQUE MISSION CAPABILITIES
      - BOOST – REBOOST – DEBOOST
      - RETRIEVE
      - VIEWING
      - SUB SATELLITE SUPPORT
      - REMOTE INSITU SERVICING & REFUELING
      - DEBRIS RETRIEVAL
  - KEY ELEMENT OF SPACE STATION
    - SERVICE UNMANNED PLATFORMS
    - RETRIEVE AND DEPLOY LARGE OBSERVATORIES
    - SUPPORT PROXIMITY OPERATIONS

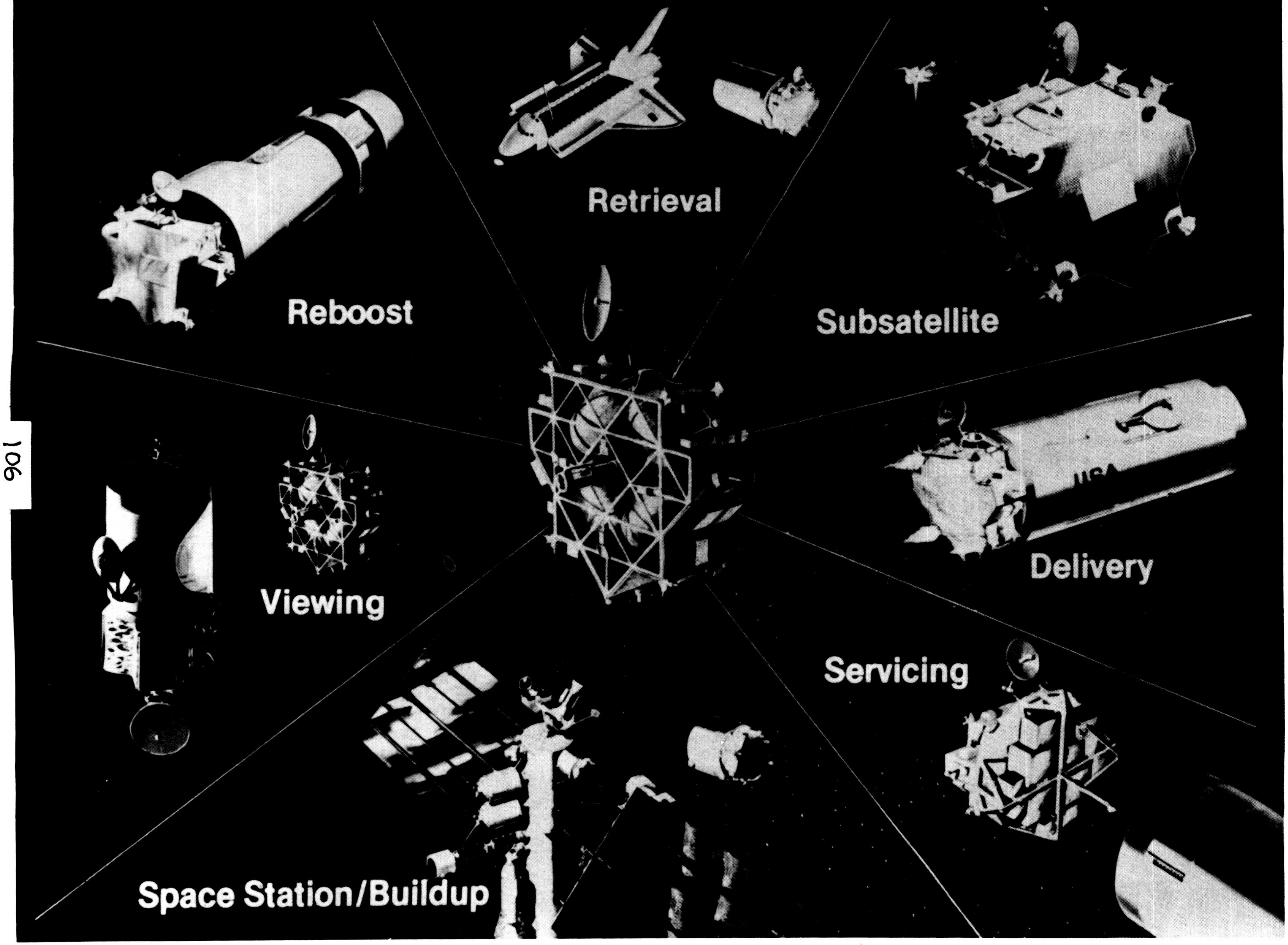
OMV OPERATIONAL MISSIONS

- The facing artistic rendering depicts several of the more obvious OMV missions/ services.

- The servicing mission as pictured represents the OMV equipped with an automated manipulator system which is capable of exchanging failed modules. Such a system could also be utilized to provide refueling umbilical connect/disconnect operations, thus enabling the servicer system to perform combined service functions.

- GEO operational capability of the OMV when combined with the OTV and debris capture functions in LEO are not shown but will be required.

OMV Operational Missions



OMV REQUIREMENTS SUMMARY

The facing chart depicts the basic OMV requirements in terms of major performance objectives, operational capabilities and hardware characteristics.

The OMV will have the capability of performing delivery, retrieval, reboost, deboost and viewing missions from both the Shuttle and Space Station.

Launch operations may be performed out of ETR and WTR and on-orbit operations may be controlled either from the ground or the Space Station. Space Station operations typically will be controlled by the Space Station operator when the OMV is operating in close proximity of the station. In either case, final docking maneuvers are performed by remote piloted modes.

Cold gas will be utilized for final docking and proximity operations maneuvers near contamination sensitive payloads.

OMV will be capable of nine (9) month storage in the event the Orbiter has to return to the ground prior to OMV return to the Orbiter.

The OMV will be designed to accommodate on-orbit maintenance and will have a design life of 10 years with ground refurbishment.

OMV payload accommodations are available in the form of structural support, power, data and command. Electrical power will be available up to 5 KWh at a peak energy rate of 1 KW. Digital data may be through put at the video rate when the TV is not being used. The capability will exist to provide the payload serial digital command data.

Mission kits in addition to the above payload services may also be utilized. Study efforts are currently being performed to define three major kit areas - remote fueling, debris retrieval, and in situ modular servicing.

5050-85

OMV REQUIREMENTS SUMMARY

- PROVIDE SPACECRAFT DELIVERY, RETRIEVAL, REBOOST, DEBOOST, AND VIEWING
- SHUTTLE BASED, (ETR & WTR), GROUND CONTROLLED
- SPACE STATION BASED, GROUND AND STATION CONTROL
- MAN IN THE LOOP CONTROL FOR FINAL OPERATIONS
- COLD GAS RCS FOR PROXIMITY OPERATIONS
- NINE MONTHS ON–ORBIT SELF–CONTAINED STORAGE
- CAPABLE OF BEING MAINTAINED ON ORBIT
- TEN YEAR LIFE WITH REFURBISHMENT
- PROVIDE LIMITED RESOURCES TO PAYLOADS
- ACCOMMODATE VARIOUS MISSION & SERVICE KITS

OMV REFERENCE DESIGN

Pictured here is the MSFC reference design which was defined prior to the initiation of the Phase B procurement competition.

The OMV project will be under A-109 procurement regulations until such time as the final Phase C/D contractor is selected. Those designs defined by the competing contractors (Vought, Martin, TRW) remain procurement sensitive.

This design is generally descriptive of the OMV required to support the design reference missions.

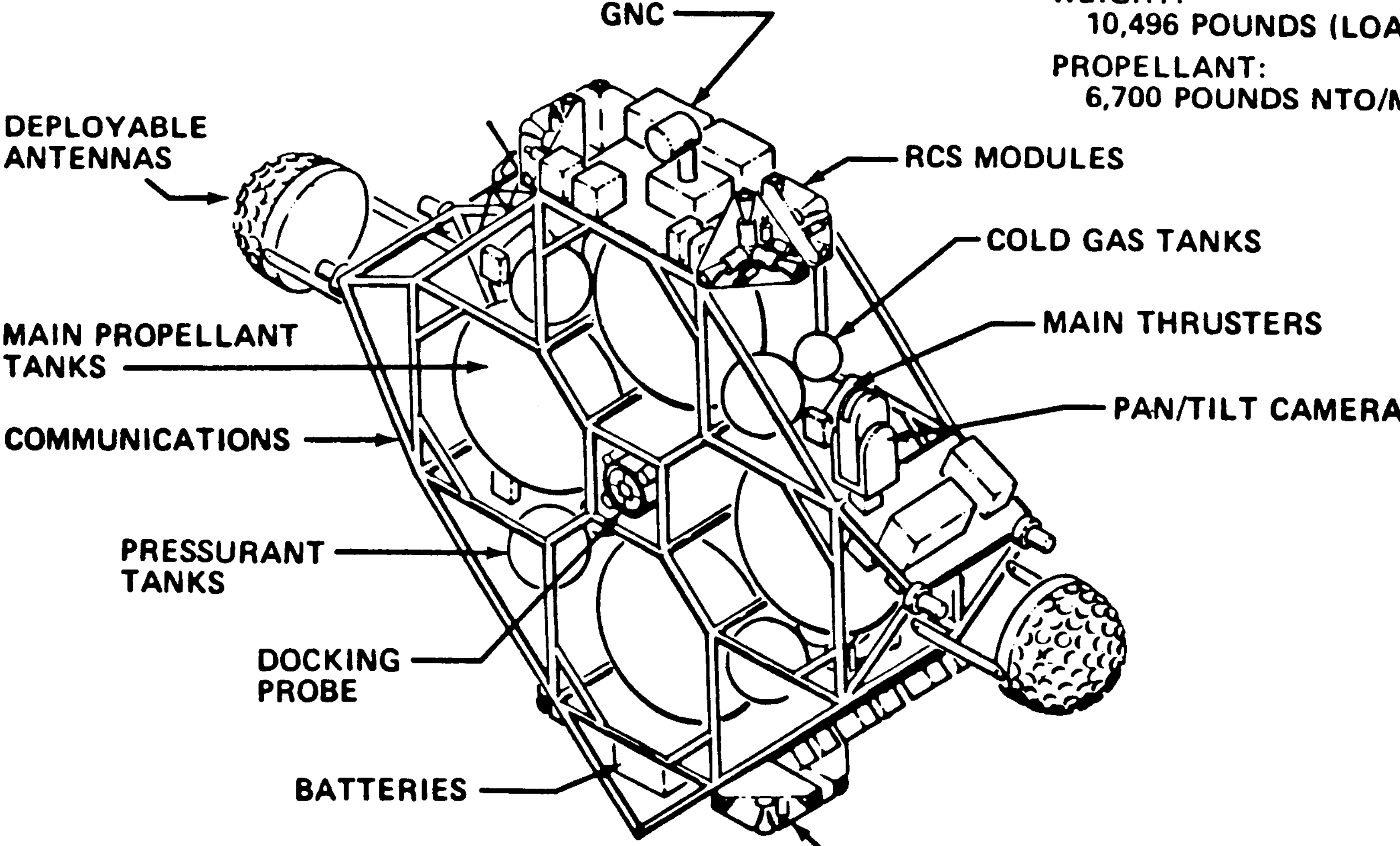
824-84

ORBITAL MANEUVERING VEHICLE (OMV) (REFERENCE DESIGN)

DIMENSIONS:
37 X 178 INCHES

WEIGHT:
10,496 POUNDS (LOADED)

PROPELLANT:
6,700 POUNDS NTO/MMH



NOTE: THERMAL CONTROL BLANKETS REMOVED FOR CLARITY

OMV LEO PERFORMANCE

This performance plot shows several alternative OMV capabilities above the Orbiter or Space Station.

The Payload Deliver curve represents the OMV performance in delivering a payload and returning without a payload.

The Payload Retrieval curve represents the performance of the OMV flying out to the payload fully fueled performing a rendezvous and dock and returning with the payload.

The Round Trip Payload curve represents the OMV capability to deliver a replacement payload to orbit and to return the original payload.

The Retrieve Redeploy performance curve represents the performance of the OMV flying out to retrieve a payload for service, returning it to the Orbiter or Space Station and redeploying the payload after service.

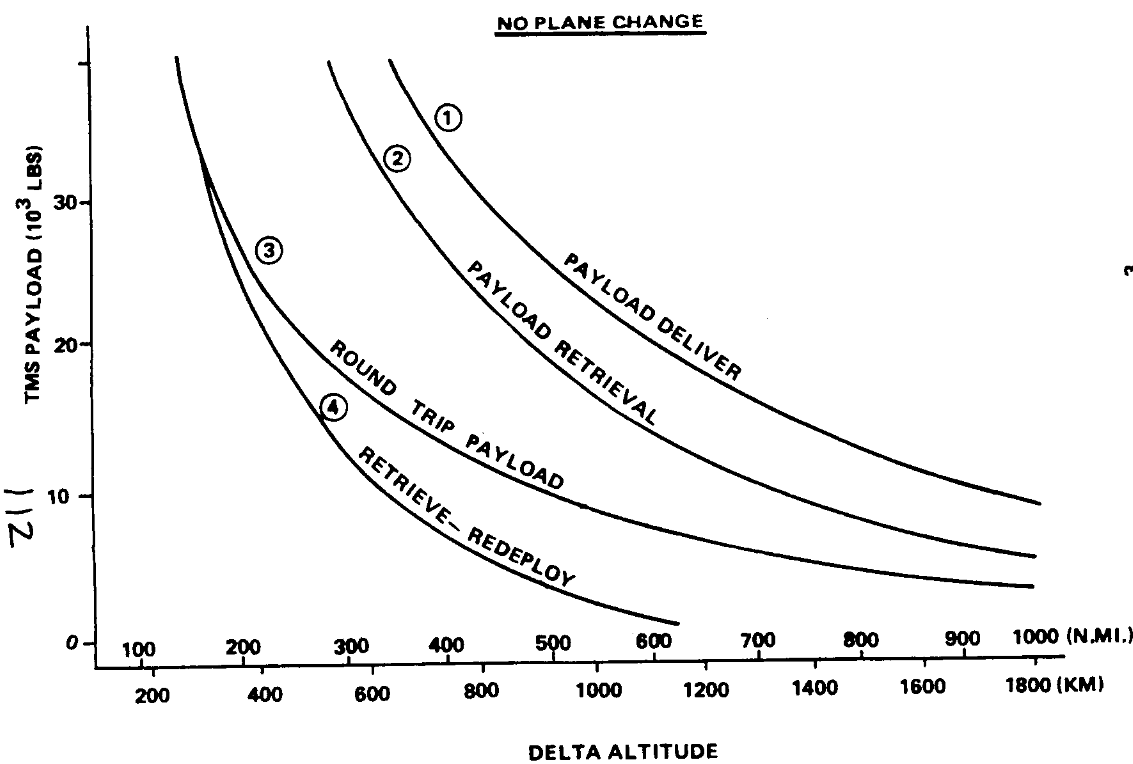
The Deployment Plane Change chart shows the OMV plane change capability combined with delivery.

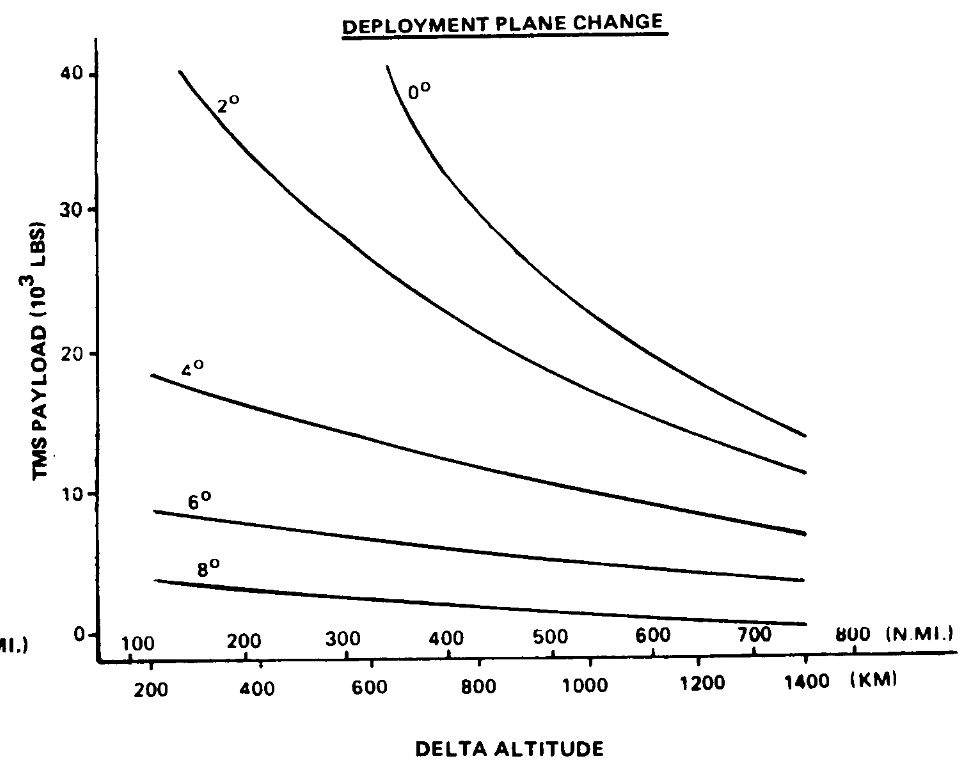
5544–83

OMV LEO PERFORMANCE

W_{PROP} = 6566 LB

I_{SP} = 285 SEC.





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OMV SUPPORT TO ST MAINTENANCE

The ST will be initially placed on orbit by the Shuttle. Subsequent repair missions may be supported by the Shuttle and OMV as described by these two scenarios.

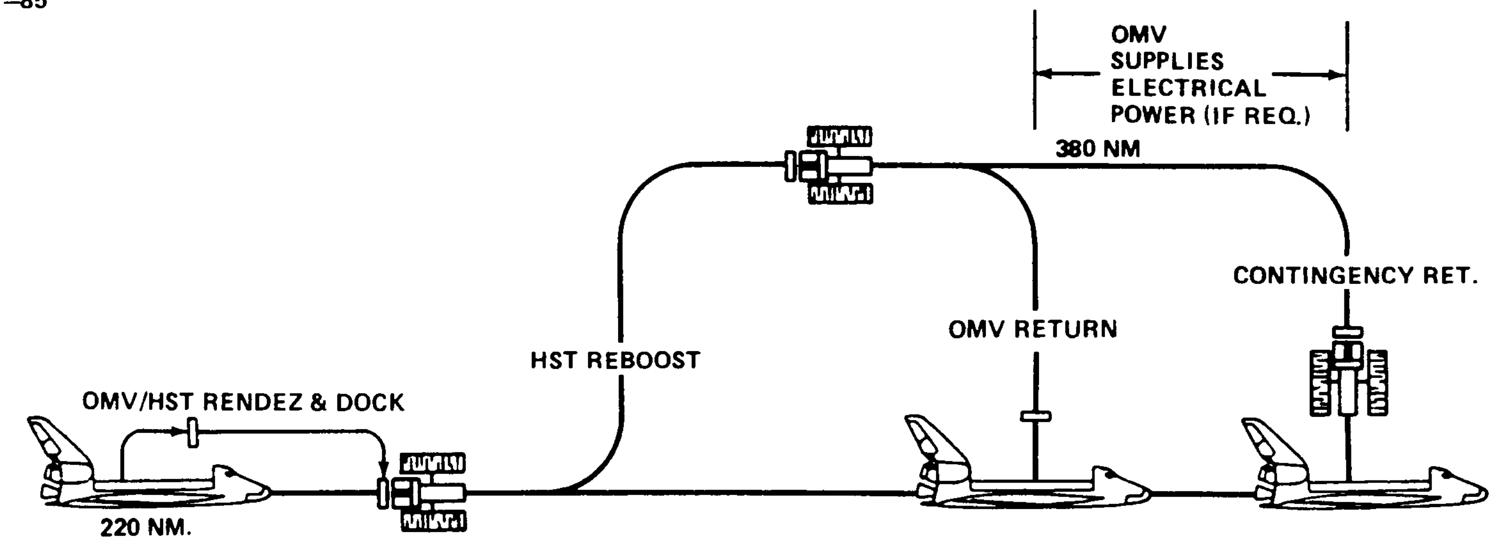
Mission Scenario #1, Initial ST reboost with Contingency Return to the STS - The STS will deliver and release the OMV in the ST orbit. The ST orbit may be as low as 220 nm or as high as 320 nm and inclined at 28.5 degrees. The OMV will dock with the ST and reboost the ST to a 380 nm operational orbit. The OMV will release the ST and remain in a standby position until the ST is checked out. The maximum time required for checkout is 2-4 hours. The OMV will retain sufficient resources for a contingency return of the ST to the STS orbit. Contingency return may be initiated at any point during reboost. Upon successful checkout of the ST, the OMV will return to the STS orbit where the STS will rendezvous and dock with the OMV.

Mission Scenario #2, ST Retrieval/Reboost with Contingency Return to the STS - The OMV will be released by the STS from a 270 nm, 28.5 degrees inclined orbit. The OMV will transfer to the ST at 270-380 nm altitude. The OMV will rendezvous and dock with the ST in the general configuration shown in the Figure and return the ST to the STS for maintenance. After return of the ST, the OMV will be placed in a standby orbit or be berthed in the STS for up to four days. After ST maintenance, the OMV will dock with and reboost the ST to a 380 nm operational orbit. The OMV will release the ST and remain in a standby position until the ST is checked out. The maximum time required for checkout is 2-4 hours. The OMV will retain sufficient resources for a contingency return of the ST to the STS orbit. Contingency return may be initiated at any point during reboost. Upon successful checkout of the ST, the OMV will return to an orbit where the STS will rendezvous and dock with the OMV.

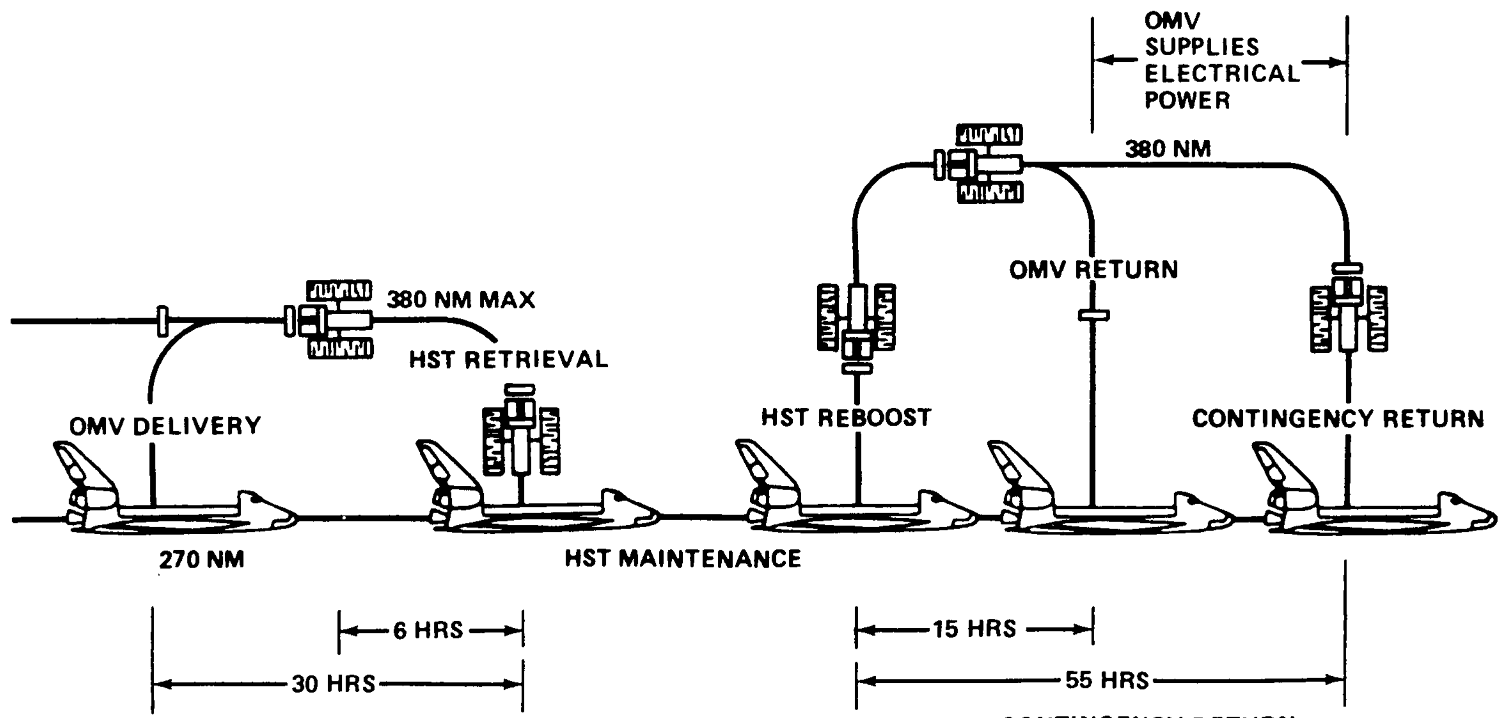
NOTE: Solar arrays are assumed to be extended at all times in both scenarios to provide power to the ST. This requirement makes it necessary to limit the g-force to .002g on the solar array structure. The OMV will be capable of performing this soft ride.

5041–85

OMV SUPPORT TO HST



MISSION SCENARIO 1 – REBOOST WITH CONTINGENCY RETURN



MISSION SCENARIO 2 – RETRIEVE, REPAIR, REBOOST, CONTINGENCY RETURN

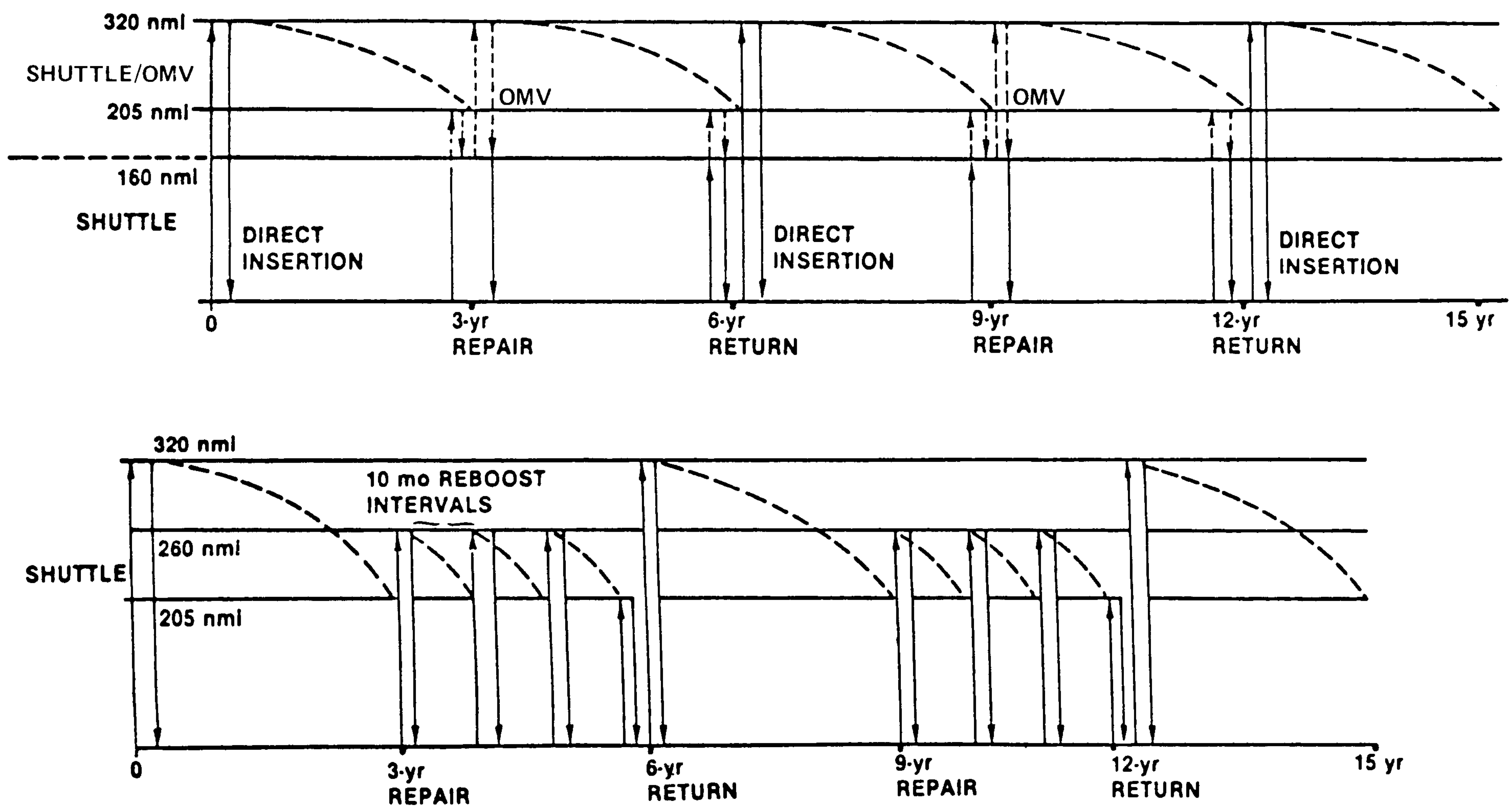
OMV SUPPORT TO AXAF

Opportunities for OMV use in support of large observatory programs are being considered in current and planned MSFC contractor and in-house studies. The facing chart displays a typical mission profile for the OMV support to an Advanced X-ray Astronomy Facility (AXAF) servicing mission. As shown, the OMV flys from the orbiter standard altitude (160 nautical miles), retrieves AXAF at 205 nautical miles (the predicted decay altitude AXAF will have obtained after three years on orbit), and brings it back to the orbiter for repair and servicing. After servicing is completed, the OMV then reboosts the AXAF back to an operational altitude of 320 nautical miles, and then returns to the orbiter for retrieval/reuse on a subsequent mission.

The benefits of using the STS-OMV combination for this mission follow: Over a 12 year period, 7 shuttle launches are involved to support the program if OMV is used. Eleven shuttle launches are required to do the same job without OMV. The reason is that the shuttle cannot reboost the AXAF as high on each revisit mission for servicing (i.e., it reboosts AXAF from an estimated 205 nautical miles to 260 nautical miles). The orbit decay rate is higher at these lower altitudes and more frequent STS launches for reboost would therefore be required. Use of the OMV saves this program the equivalent of 4 dedicated STS launches over the 12 year period, resulting in a substantial operations benefit and cost savings to the project.

5043–85

AXAF MISSION SCENARIOS



P2-3311-16A

ORBITAL SERVICING STUDY OVERVIEW

The Orbital Maneuvering Vehicle can provide support to a smart front-end kit, or automated servicer system, capable of remotely controlled maintenance of spacecraft designed to utilize this capability. These in situ servicing functions may include modular exchange of subsystem and experiment packages, fluid transfer and resupply, replenishment of raw material modules and pickup of processed product modules.

An extensive study of satellite servicing has been conducted by MSFC. The major elements of this study are depicted here. The study consisted of comparing the cost effectiveness of expandable spacecraft programs versus maintenance and repair after return and retrieval to earth versus visiting spacecraft systems. Two subalternatives of visiting systems were EVA maintenance at the Orbiter versus in situ servicing. The study results show that on-orbit maintenance is the most cost effective mode. The on-orbit servicer defined in these studies utilized a module exchange approach which greatly simplifies the servicer system and reduces operational complexity of the repair process. It was found that most maintenance tasks were reduceable to a simple remove and replace operation. This module exchange approach implies specifically designed spacecraft; however, the design constraints, impacts, and penalties were not considered serious.

5048–85

ORBITAL SERVICING STUDY – OVERVIEW

MAINTENANCE APPROACHES CONSIDERED

- EXPENDABLE
- ON–GROUND
- VISITING SYSTEMS
  - EVA
  - SRMS
  - ON–ORBIT SERVICERS
    - PIVOTING ARM
    - GENERAL PURPOSE

CONCEPTS

SELECTED

MANIPULATOR

EXPENDABLE

GROUND REFURBISHABLE

ON–ORBIT MAINTENANCE

MODES

- ON–ORBIT MAINTENANCE IS MOST COST EFFECTIVE MODE
- ON–ORBIT SERVICERS ARE MORE VERSATILE
  – USEFUL AT ORBITER, AT SPACE STATION, ON OMV, AT GEO WITH APPROPRIATE CARRIER VEHICLE
- PIVOTING ARM USES MODULE EXCHANGE APPROACH
  – EMPHASIZES OPERATIONAL SIMPLICITY
  – MOST MAINTENANCE TASKS REDUCIBLE TO REMOVE AND REPLACE
- MODULE EXCHANGE IMPLIES SPECIALLY DESIGNED SPACECRAFT
  – CONSTRAINTS AND PENALTIES NOT SERIOUS

REMOTE SERVICING APPLICATION AND BENEFITS

The applications and benefits of on-orbit servicing are shown here. A servicing capability integrated with the mission capabilities of the orbital maneuvering vehicle will extend the access envelope for which servicing can be provided. Low earth orbit altitudes up to 1,000 nautical miles will be within the reach of this servicing capability. The semi-automated teleoperational approach for delivering this servicing function will reduce the need for manned modules, will reduce the need for retrieval to the Space Station or the Orbiter for servicing, and will reduce the cost associated with extra vehicular activities. The on-orbit servicing approach will prove to be cost effective for many, but not all, satellite servicing functions, depending on the spacecraft orbit and maintenance requirements. Retrieval and EVA will continue to be the most cost effective answer for many spacecraft.

REMOTE SERVICING APPLICATION BENEFITS

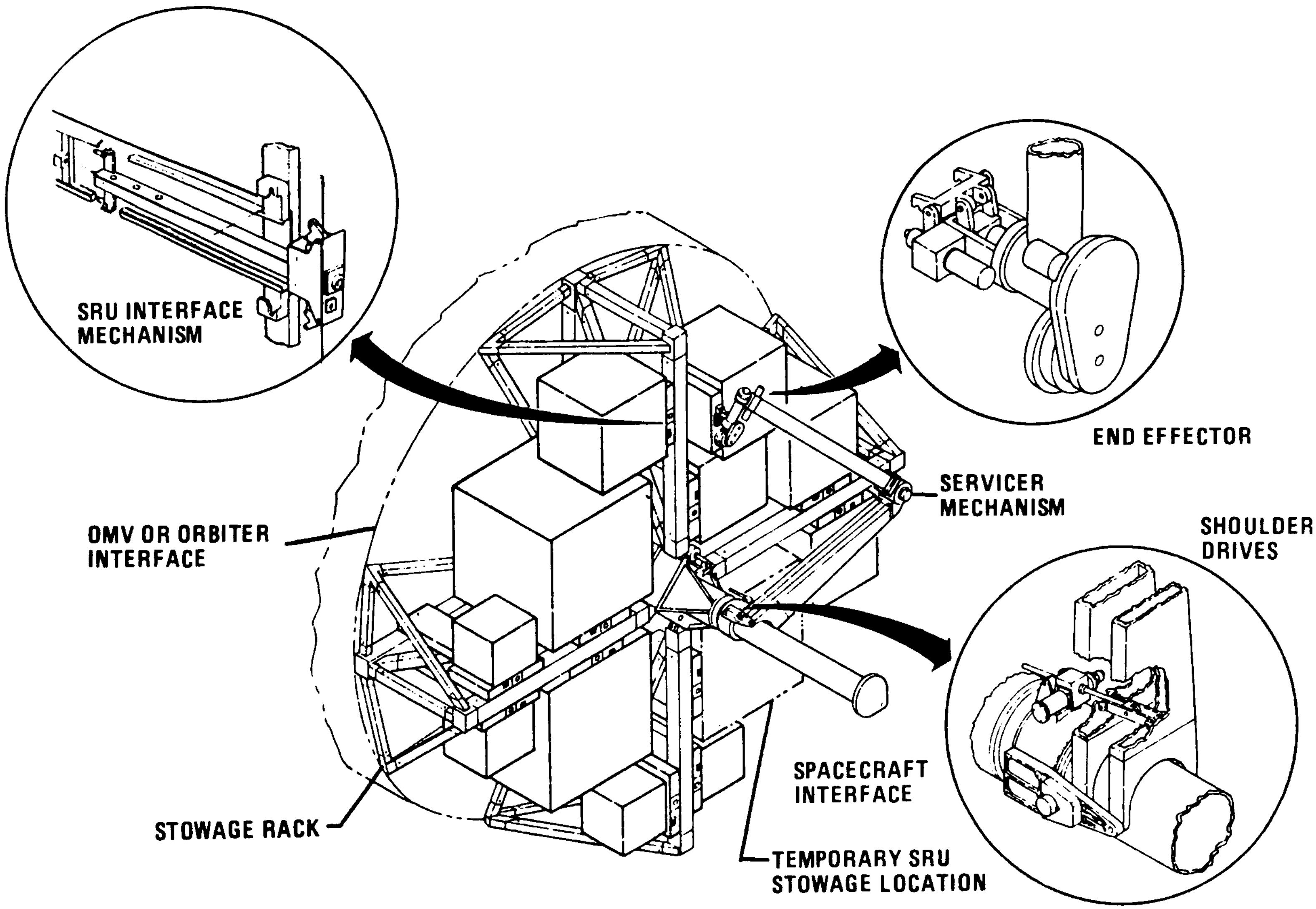
- BASIS
  - EXTEND ORBITAL ACCESS
  - REDUCE NEED FOR MANNED MODULES
  - REDUCE EVA RELATED DEPENDANCE
  - ASSIST EXTENSION OF TERRESTRIAL AUTOMATION TO SPACE
- APPLICABLE ORBITS
  - LOW EARTH ORBIT (LEO)
    (INTERMEDIATE ALTITUDE AND INCLINATION CHANGES)
  - GEOSYNCHRONOUS EARTH ORBIT (GEO)
    (USING AN ORBITAL TRANSFER VEHICLE)
- REPRESENTATIVE FUNCTIONS
  - REPAIR
  - RESUPPLY
  - UPGRADING
  - PRODUCT RETURN
- REPRESENTATIVE PROGRAMS
  - SPACE TRANSPORTATION SYSTEM
  - SPACE STATION
  - FREE FLYING PLATFORMS
  - LEO SATELLITES
  - GEO SATELLITES

SERVICER-FLIGHT UNIT

The servicer and storage rack design, which resulted from the servicer system studies, is shown here. The storage rack consists of a tubular truss design that mounts on the front face of the Orbital Maneuvering Vehicle. Modules of various sizes and shapes can be arranged on the storage rack. The interface mechanism for attachment of the modules to the storage rack is the same mechanism that is utilized for attaching the module to the operational spacecraft. The servicer utilizes a docking probe for making a firm and hard mechanical interface with the operational spacecraft. A particular docking probe design is depicted in the picture, but the design can vary so that it is compatible with the spacecraft to be serviced. A pivoting arm form of servicer mechanism is used to remove modules from the operational spacecraft and place them on the storage rack. The arm is also utilized for the module replace function. An end effector that is compatible with the module attachment mechanism is shown. This end effector contains both a grip capability and a power takeoff capability for loosening and tightening the module fasteners. The servicer design and interface requirements are compatible with the Orbital Maneuvering Vehicle payload support capability.

5048–85

ON–ORBIT SERVICER – FLIGHT UNIT



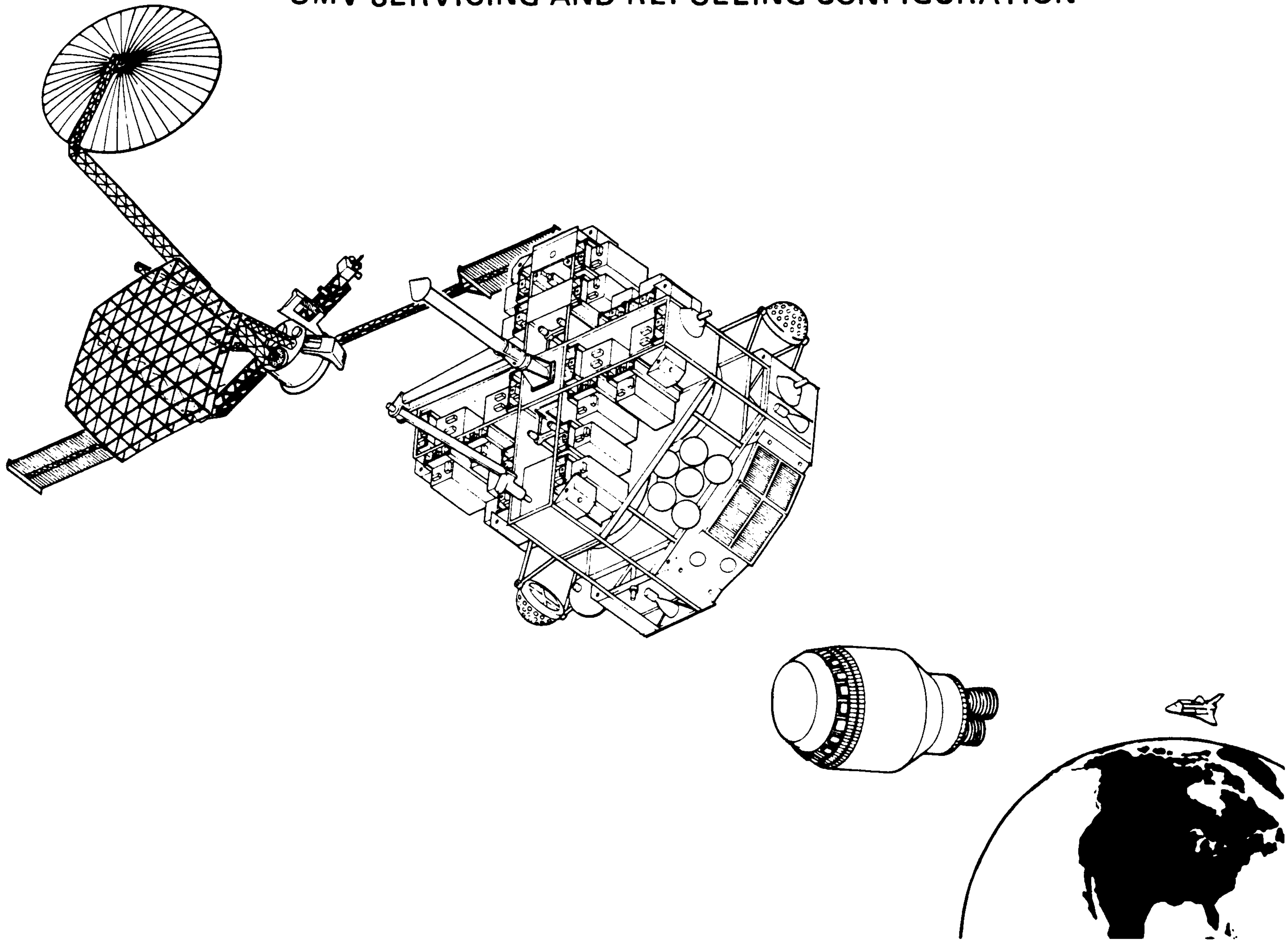
PLATFORM AUTOMATED SERVICING

Automated maintenance and resupply can be performed in high earth orbits by a Servicer System being delivered by the OMV. Similar functions can be performed at GEO with the Servicer being delivered with an OMV/OTV vehicle combination as depicted here.

Engineering Test Unit (ETU) simulation systems which demonstrate the technology of modular automated servicing are currently being tested and evaluated at MSFC. Efforts are currently being directed toward ground simulation and definition of orbital flight experiments to demonstrate the system capability to candidate users. Development of standardized interfaces between the OMV and future spacecraft will permit the utilization of remote servicing capabilities now being defined with acceptable design impact on the payload.

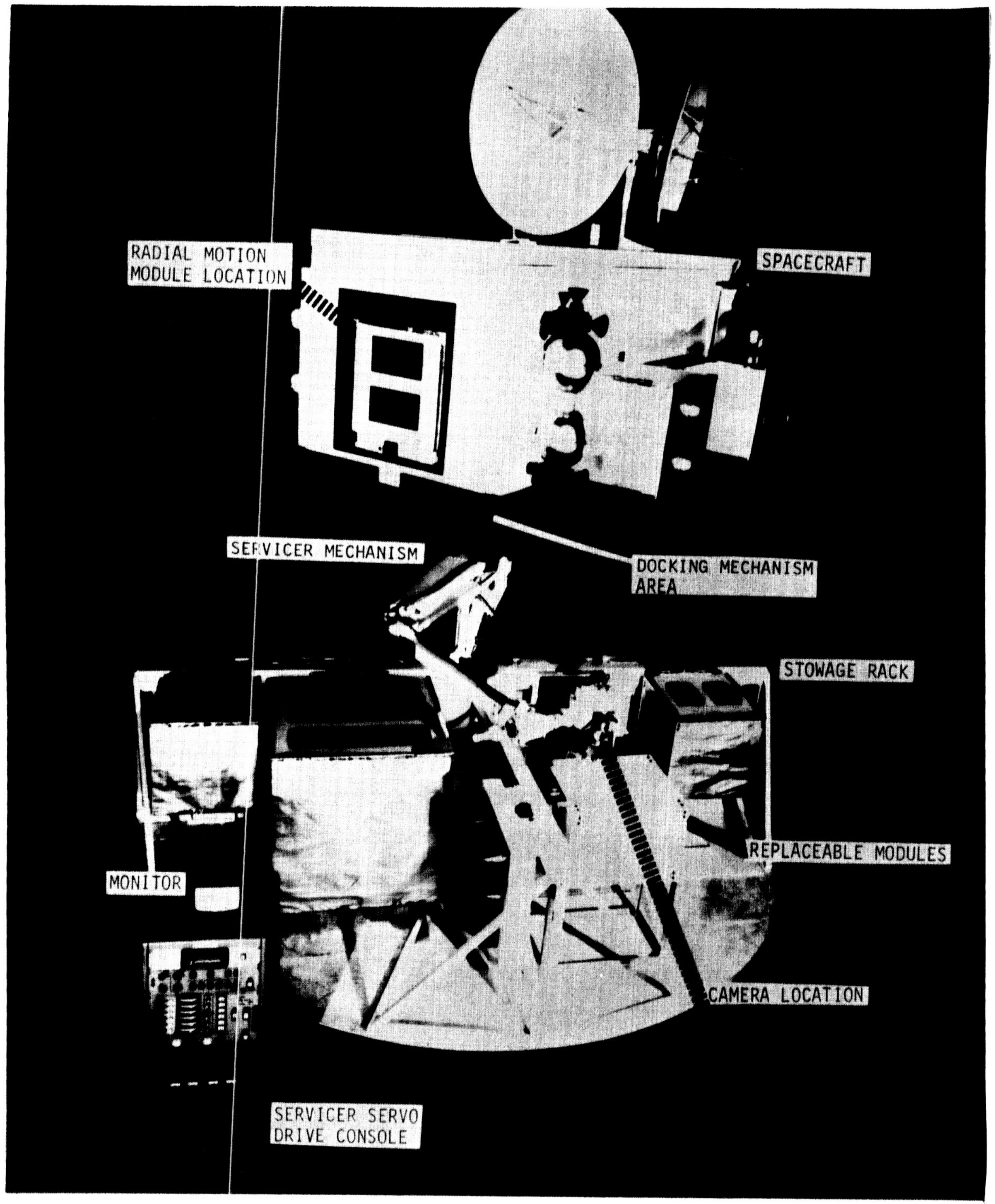
5042–85

SPACE PLATFORM
OMV SERVICING AND REFUELING CONFIGURATION



SERVICER ENGINEERING TEST UNIT (ETU)

The servicer system design has evolved into an Engineering Test Unit (ETU) which is being utilized to support simulated servicing activities at Marshall Space Flight Center. The ETU includes a mockup of an operational spacecraft, module attachment mechanisms, a breadboard of the storage rack, and an operational servicer arm mechanism that is utilized to transfer modules between the spacecraft and the storage rack. A control panel is also included as part of the ETU for controlling the system. The servicer arm is capable of being directed by an automated computer control program to perform module replacements. The arm is directed to move to the module location, effect firm mechanical interface with the module, release the module fastener, transport the module to the location for its placement, tighten the fastener to engage the module in the surrounding structure, demate the end effector from the module and move the arm to a neutral position. This system has been operated several hundred times to demonstrate its repeatability and reliability.



SPACECRAFT, SERVICER MECHANISM, AND STORAGE RACK

SERVICEABLE COMMUNICATIONS SPACECRAFT

The modular maintenance concept is highly dependent upon the compatible design of modular spacecraft.

There is some reluctance by designers to design for 100% maintenance capability which becomes possible when all active components are housed in removeable modules. Part of this reluctance stems from the fear of increased cost and weight. Our studies indicated that such designs could be developed for a nominal increase of 4% in design and development cost with a corresponding 8% increase in hardware cost.

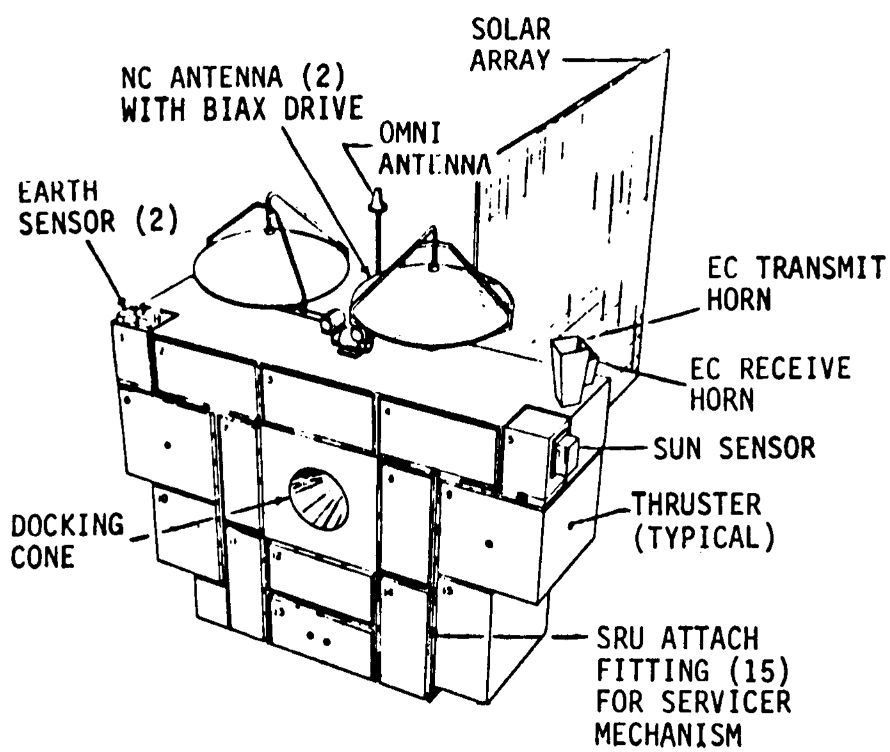
The above costs to enable modular maintenance are dwarfed by the potential savings to the project which varies from 20 to 35%.

Several modular spacecraft designs were completed in our studies. These are well represented by the spacecraft which follow.

The Serviceable Communications Spacecraft shown on the facing page is one of three serviceable spacecraft designs prepared by TRW. It is an excellent representative of the form all geosynchronous communications satellites might take. It is a single tier and is box shaped. All modules are removed axially. In this configuration, a single solar array mounted opposite the docking port is used. An advantage of the configuration shown is that the exposed faces of the modules, or orbit replaceable units (ORUs), see very little of the sun and thus can be used to radiate heat out of the modules.

A breakdown of the spacecraft mass properties shows that 1,920 lb, or 81% of the total spacecraft weight, is space-replaceable. The major items that are not serviced are the basic structure, the solar array (the solar array drive is replaceable), the narrow-coverage antennas and their biax drives, the horn antennas, the omni antenna, and the shunt element assembly.

DEFENSE SUPPORT COMMUNICATIONS SPACECRAFT II



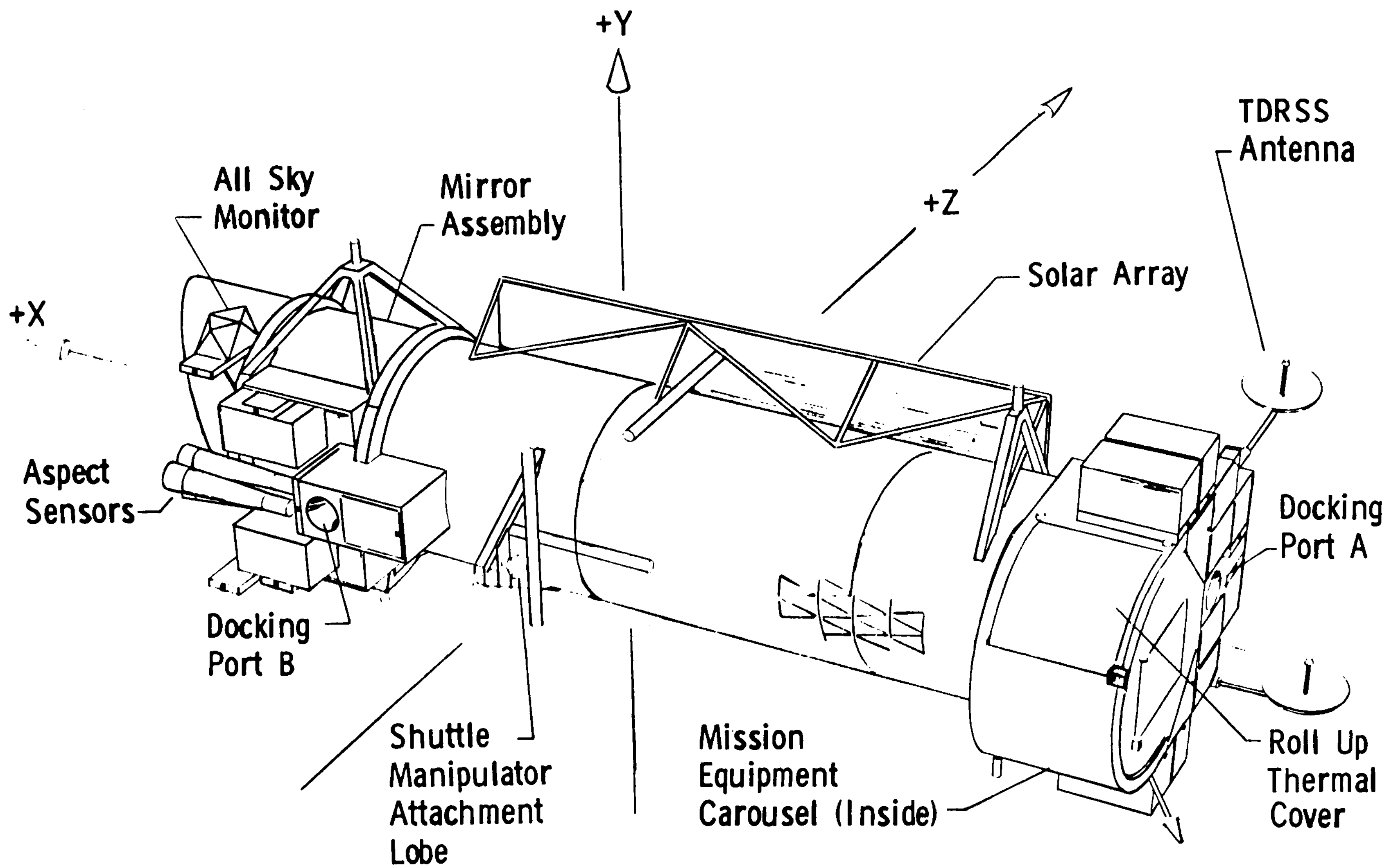
- GEOSYNCHRONOUS ORBIT
- BOX SHAPE
- 99 IN. X 128 IN. X 40 IN.
- WEB STRUCTURE
- SINGLE CENTRAL DOCKING
- SINGLE TIER
- AXIAL MODULE REMOVAL
- 15 MODULES
- LARGEST--40 IN. X 40 IN. X 32 IN.
- HEAVIEST--444 LBS
- MINIMUM REACH--23 IN.
- MAXIMUM REACH--72 IN.

SERVICEABLE CHARACTERISTIC LARGE OBSERVATORY (CLO)

The Characteristic Large Observatory (CLO) spacecraft design was prepared by TRW. It represents three classes of large low-earth orbit observatories - X-ray, stellar, and solar. The stellar and solar versions were addressed in terms of their unique mission equipment. The CLO incorporates two docking ports, one aft and one forward and to the side, with the modules at each docking port arranged in a single tier. The second docking is required because the aspect sensors must be mounted at the mirror assembly and because there were too many modules to be mounted in a single tier. There are several outsize mission equipment modules, but these can be mounted in the nominal stowage rack. Roll-up thermal covers can be rolled back by the servicer end effector and then the mission equipment modules can be removed from the carousel.

The CLO represents one of the most complex spacecraft examined for serviceability. By having the servicer system designers and the spacecraft designers interact, it was possible to come up with a good serviceable CLO design in a short time.

SERVICEABLE CHARACTERISTIC LARGE OBSERVATORY



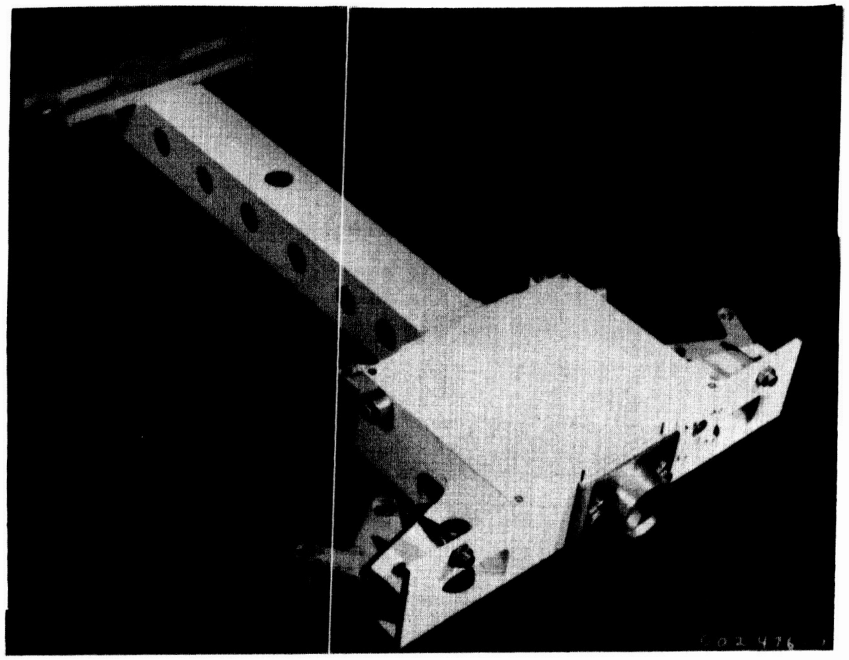
SIDE MOUNTING INTERFACE MECHANISM

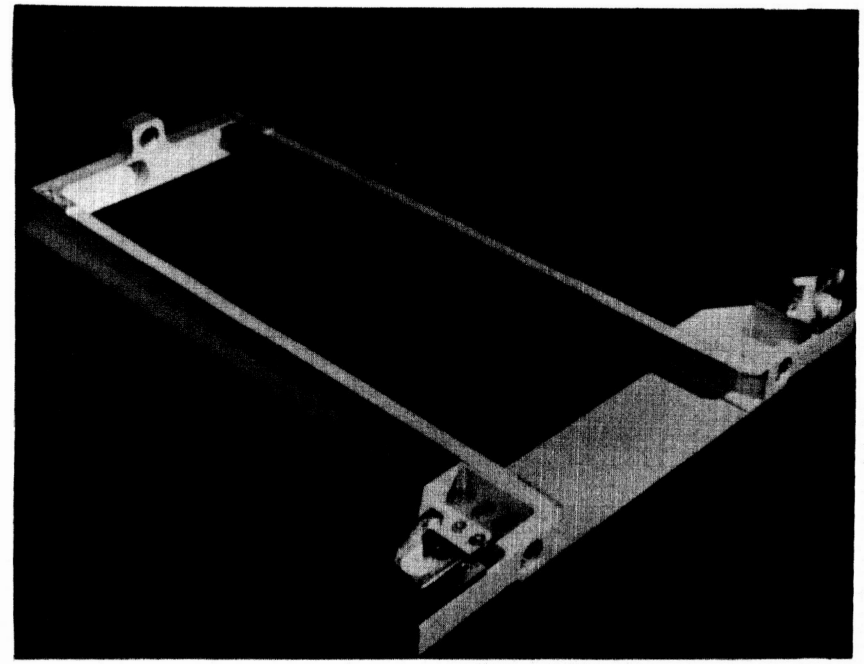
Key to the ability to perform modular maintenance is the module attachment mechanism. Two types were designed and built - a bottom mount and a side mount which is shown here. The module used in demonstrations was a 24 inch cube configured for minimum weight with adequate strength. The mechanism uses a three point, nonredundant, attachment system so spacecraft thermal and structural loads do not pass through the module. The bell crank linkage is driven via a worm and gear powered by a motor on the end effector. A spring-loaded self-aligning mechanical drive accomplishes the mechanical interface. The linkage starts engagement with a low force that gradually increases to 200 lb as the links approach an over-center position. Total travel is 1-3/4 inches.

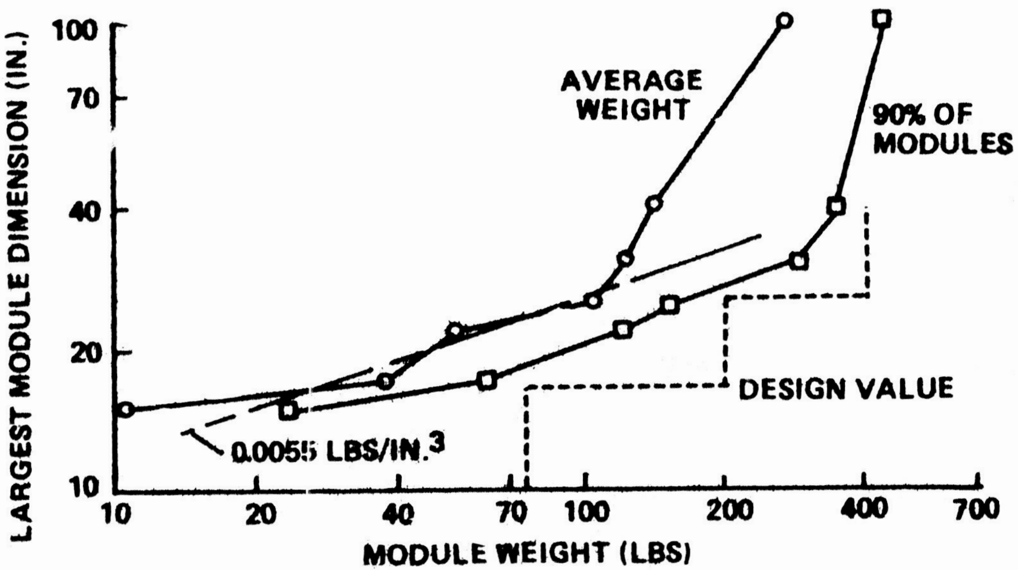
The attach cone has a ± 3/4 inch capture volume, while the baseplate-to-guide capture volume is ± 1/2 inch. This large capture volume is gradually reduced during engagement by the shape of the guide rails to less than 0.001 in. at final alignment. The tolerance is less than 0.005 in. during electrical connector engagement. Status indicators were provided on each set of guides for each type of interface mechanism. Cams and microswitches were used for the "ready" and "unlatched" indications and mating of the electrical connector for the "latched" signal.

Interface mechanisms could be developed as a two-part kit in perhaps three sizes. These standard interface mechanisms could be made available to spacecraft designers. Each designer could then make his choice within his own set of design and economic constraints. The graph on the facing page is a histogram from data on 683 modules from 30 different serviceable spacecraft. The recommended interface mechanism standard sizes thus became 17 in., 26 in., and 40 in. These correspond to modules no larger than a cube of the indicated dimension. The recommended corresponding module weight limits are shown on the graph.

SIDE INTERFACE MECHANISM







MMS GROUND SERVICING DEMONSTRATION

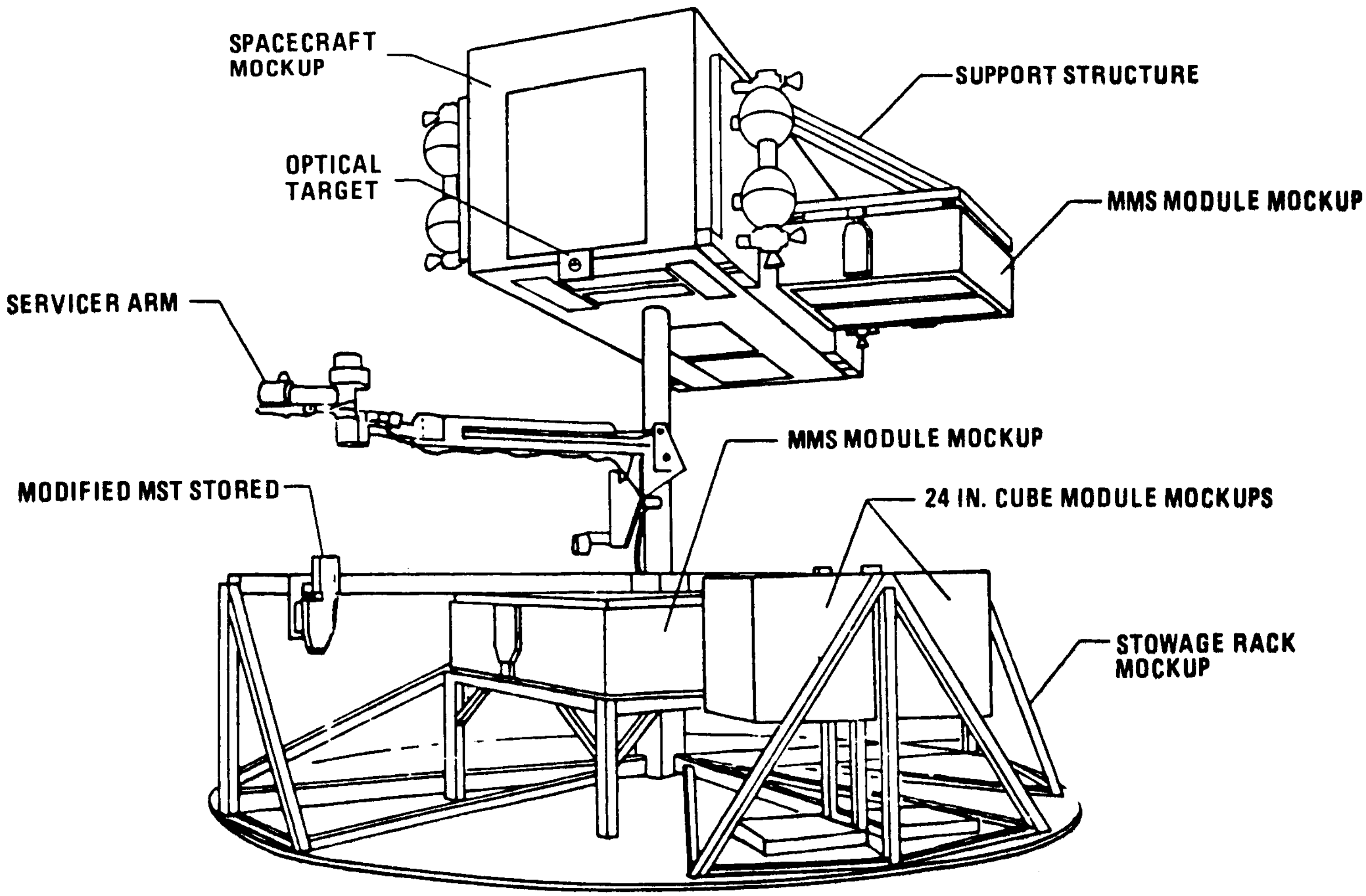
Ground demonstrations of generic modular servicing have been conducted at MSFC for a number of years. The use of the term "generic" here means that the servicer system was utilized to exchange modules expressly designed to be powered by the servicer system interface.

It has been suggested that the system should be able to accept the challenge of exchanging modules which were not designed with the servicer in mind. Rather, the servicer should be able to perform some task otherwise assigned to EVA.

This challenge has recently been successfully demonstrated by the adaptation of the Multi-Mission Spacecraft (MMS) and the Module Service Tool (MST) to the servicer system. In this demonstration a lightweight version of the MST, which was developed by GSFC supported by Fairchild, was integrated with the servicer end effector. In this configuration, the servicer readily removed and replaced a representative MMS module with the servicer being directed by an automatic computer aided control mode. This activity clearly demonstrated the servicer adaptability to a wider range of tasks.

5046-85

MMS GROUND SERVICING DEMONSTRATION



SHUTTLE FLIGHT EXPERIMENT

- MMS SERVICING -

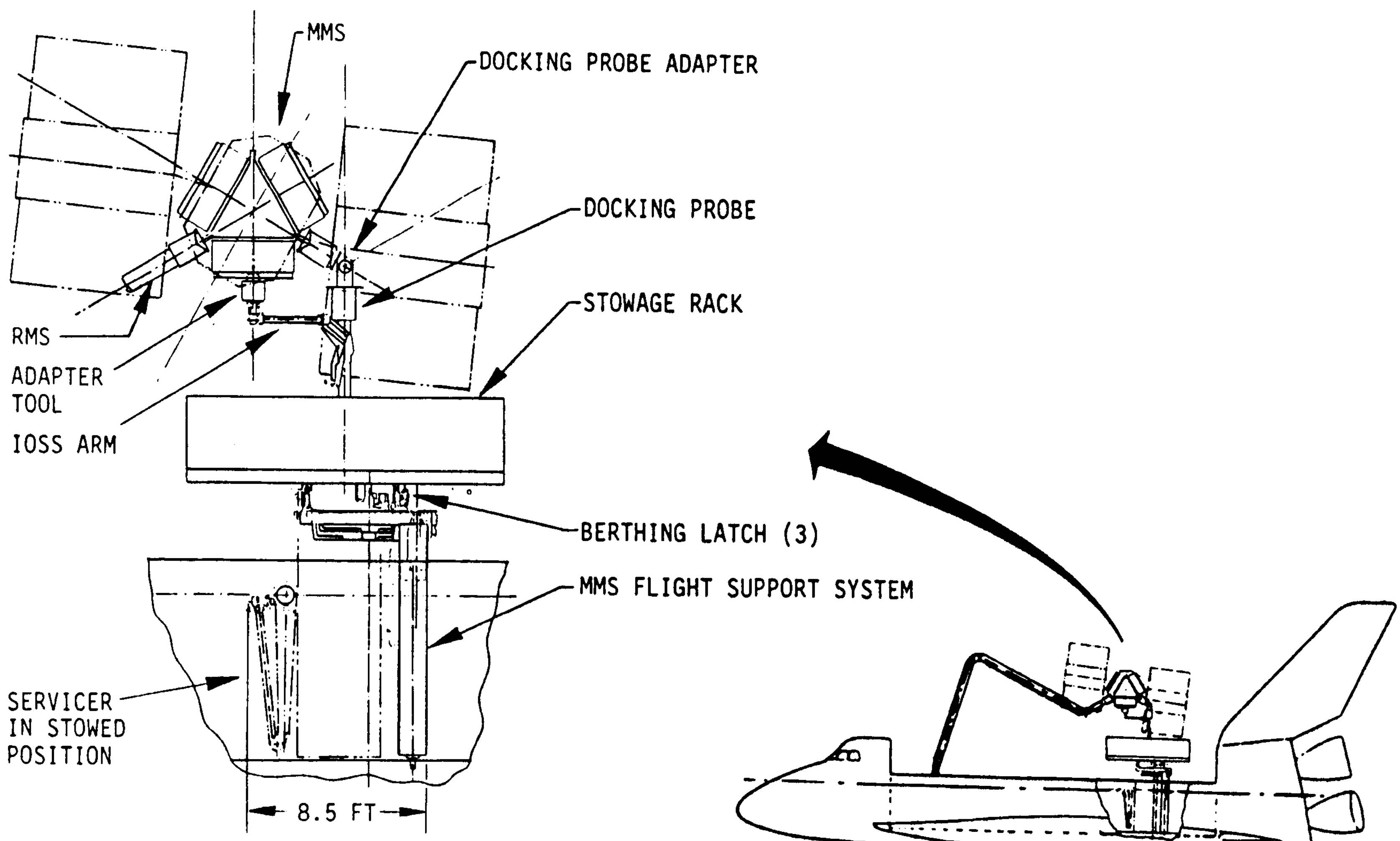
What is needed now is an on-orbit demonstration. GSFC and MSFC have entered into a cooperative endeavor in which it is proposed that an in bay shuttle experiment be flown. In this experiment/demonstration the servicer will be used to perform MMS module exchanges utilizing the MST. This flight experiment configuration is pictured on the facing page.

In this demonstration the MMS and the Servicer System will be flown to orbit supported by the Flight Support System (FSS). On-orbit the RMS will represent/simulate the OMV free flying function and will dock the MMS and the Servicer together. Subsequent to this docking maneuver, the Servicer System will be directed to exchange the three MMS modules. The system will be controlled by an operator on the aft flight deck utilizing an operator initiated computer control mode.

The successful completion of the above demonstrations should be positive proof that this servicing concept is ready to move into the next phase of development of a kit to augment OMV capability to perform remote maintenance.

5045–85

– SHUTTLE FLIGHT EXPERIMENT –
– MMS SERVICING –



SERVICER SYSTEM DEVELOPMENT

One-G Demo

This schedule reflects the time phasing of the ongoing demonstration effort at MSFC which is being supported by GSFC, Martin and Fairchild.

It is currently planned to expand the ground demonstrations next year to include fluid umbilical disconnect and connect routines including flex line management. In addition special emphasis will be placed on the analysis of requirements for and characterization of docking alignment sensors.

Depending on available funding, special efforts may further investigate and demonstrate special interface mechanisms and connectors to enable remote battery exchange.

Cargo Bay Demo

This schedule assumes funding for the in bay demonstration in FY 86 which is at best very uncertain at this time. In the absence of such funding the definitions and trades relative to the system requirements will be worked at a diminished level under the continuing study and demonstration efforts.

It is timely that the flight demonstration be funded if such a demonstration is to be accomplished prior to the Critical Design Review (CDR) of the Servicer System OMV Kit development. Such phasing would permit the integration of demonstration flight data into the final phases of the Servicer Kit development.

OMV Servicer Kit Development

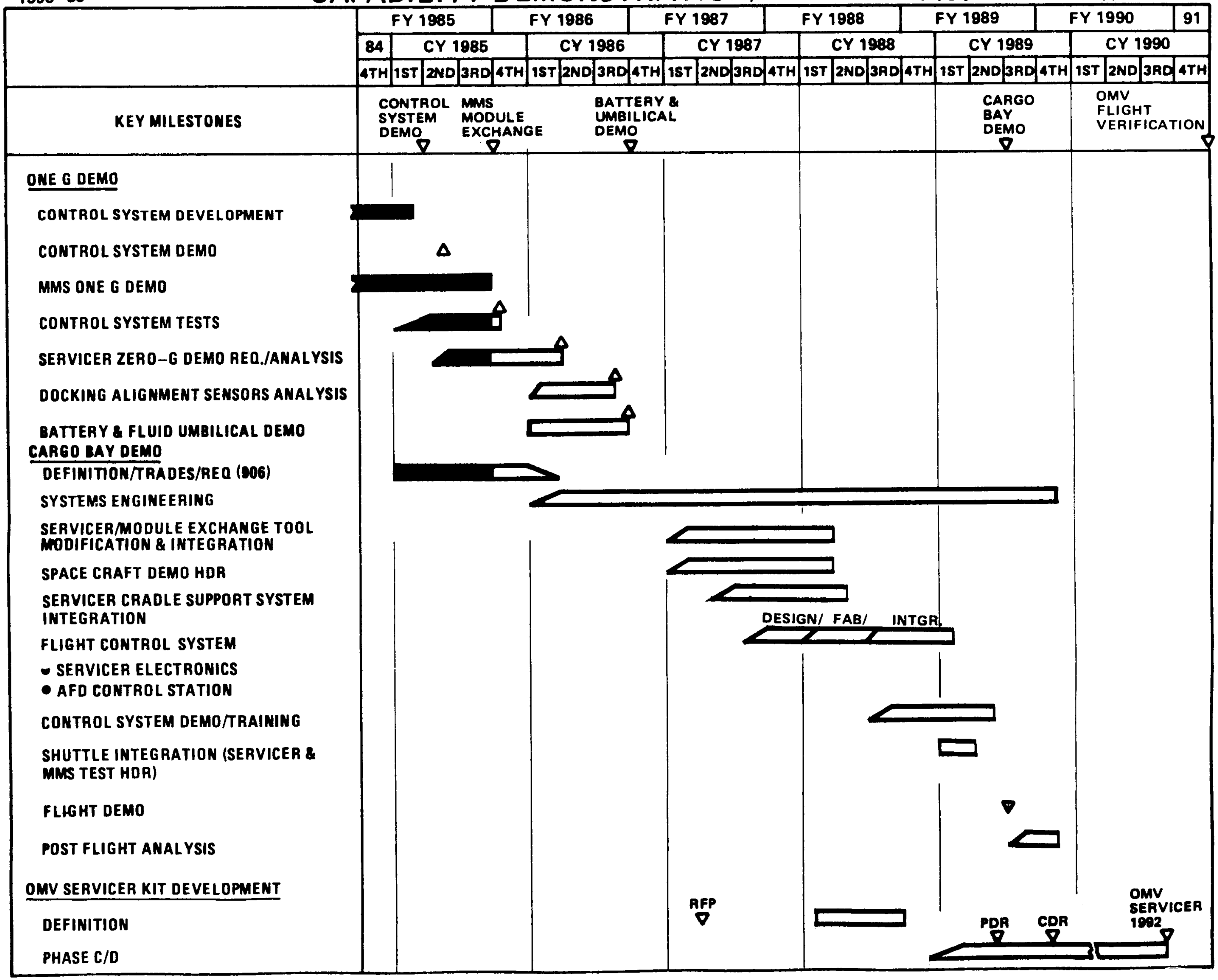
The phasing of the OMV add on Servicer Kit is shown as initiating in the 1987/88 time period. This phasing would permit the delivery of a flight type servicer in 1992 approximately one and one half years after the OMV Flight Verification.

Such phasing would provide the OMV with a Servicing System capability in time to support the Space Station and the Polar Platform.

1593–85

SERVICER SYSTEM
CAPABILITY DEMONSTRATION/DEVELOPMENT

PF–14/J. TURNER
PP02/K. TURNER
SEPT. 1985



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CONCLUDING REMARKS

The OMV and the associated services which it makes possible will greatly enhance the STS program operating capabilities.

A vast array of satellite services will come on line which will greatly enhance mission flexibility available to mission planners.

Satellite servicing, which has been successfully performed by EVA at the Shuttle, will now extend beyond man's reach but but remain under his remote control. This capability will project a profound impact on the Space Station era satellites.

Timely initiation of the OMV project and the servicing capabilities which it projects will insure these capabilities will be on line in time to support the Space Station.

5044–85

CONCLUDING REMARKS

- THE OMV, WITH MODULAR GROWTH POTENTIAL AND MULTI–PURPOSE MISSION KIT SUPPORT FEATURES, WILL COST EFFECTIVELY COMPLEMENT AND EXPAND THE PAYLOAD SERVICES OFFERED BY THE SPACE TRANSPORTATION SYSTEM

- OPERATING IN CONJUNCTION WITH PLANNED UPPER STAGES, THE OMV OFFERS A WIDE RANGE OF SERVICES TO P/L'S, OUT TO & INCLUDING GEOSTATIONARY ORBITS

  – PROPULSION APPLICATIONS
  – SERVICING APPLICATIONS

- IN THE SPACE STATION ERA, OMV WILL PROVIDE A NUMBER OF PAYLOAD AND ORBITAL SUPPORT FUNCTIONS ESSENTIAL TO OVERALL SPACE STATION MISSION OBJECTIVES

- IF AUTHORIZED AS AN FY86 START, INITIAL OMV CAPABILITIES CAN BE BROUGHT ON LINE IN THE LATE '90 – EARLY '91 TIME PERIOD

THE CRYOGENIC FLUID MANAGEMENT FACILITY

FLIGHT EXPERIMENT PROJECT

Presented to:
Satellite Services Workshop II
L.B. Johnson Space Center, Houston, TX
November 6-8, 1985

Presented by:
David M. DeFelice
NASA LeRC

CRYOGENIC FLUID MANAGEMENT FACILITY

AGENDA

- CFMF BROAD OBJECTIVE
- CFMF TECHNICAL OBJECTIVES
- CFMF APPROACH
- CFMF SYSTEM DESCRIPTION
- SELECTED CFMF TECHNOLOGIES
- PROJECT STATUS

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CRYOGENIC FLUID MANAGEMENT FACILITY

CFMF BROAD OBJECTIVE

PROVIDE TECHNOLOGY TO ENABLE DESIGN OF EFFICIENT SYSTEMS FOR MANAGING SUBCRITICAL CRYOGENIC FLUIDS IN THE SPACE ENVIRONMENT INCLUDING STORAGE, SUPPLY, AND TRANSFER

CRYOGENIC FLUID MANAGEMENT FACILITY

CFMF TECHNICAL OBJECTIVES

- LIQUID STORAGE
  - THERMAL PROTECTION SYSTEM PERFORMANCE
    - STORAGE TANK (DEWAR)
    - RECEIVER TANK (THICK MLI)
  - PRESSURE/STRATIFICATION CONTROL
    - PERFORMANCE OF THERMODYNAMIC VENTS

- LIQUID SUPPLY
  - FLUID ACQUISITIONING/POSITIONING
    - STORAGE TANK TOTAL COMMUNICATION LIQUID ACQUISITION DEVICE (LAD) PERFORMANCE
    - RECEIVER TANK PARTIAL COMMUNICATION LIQUID ACQUISITION DEVICE (LAD) PERFORMANCE
    - RECEIVER TANK RCS SETTLING & OUTFLOW
  - PRESSURIZATION SYSTEM PERFORMANCE
    - AUTOGENOUS
    - HELIUM

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CRYOGENIC FLUID MANAGEMENT FACILITY

CFMF TECHNICAL OBJECTIVES (CONT'D)

* LIQUID TRANSFER
  - THERMAL CONDITION OF OUTFLOW
  - LIQUID QUANTITY GAGING
  - MASS FLOW METERING (TWO-PHASE DETECTION)
  - CHILLDOWN
    - TRANSFER LINE
    - RECEIVER TANK
  - RECEIVER TANK NO-VENT FILL
  - SUPPLY TANK AND ACQUISITION DEVICE REFILL
  - VENTING OF NONCONDENSIBLE HELIUM
  - RECEIVER TANK REFILL

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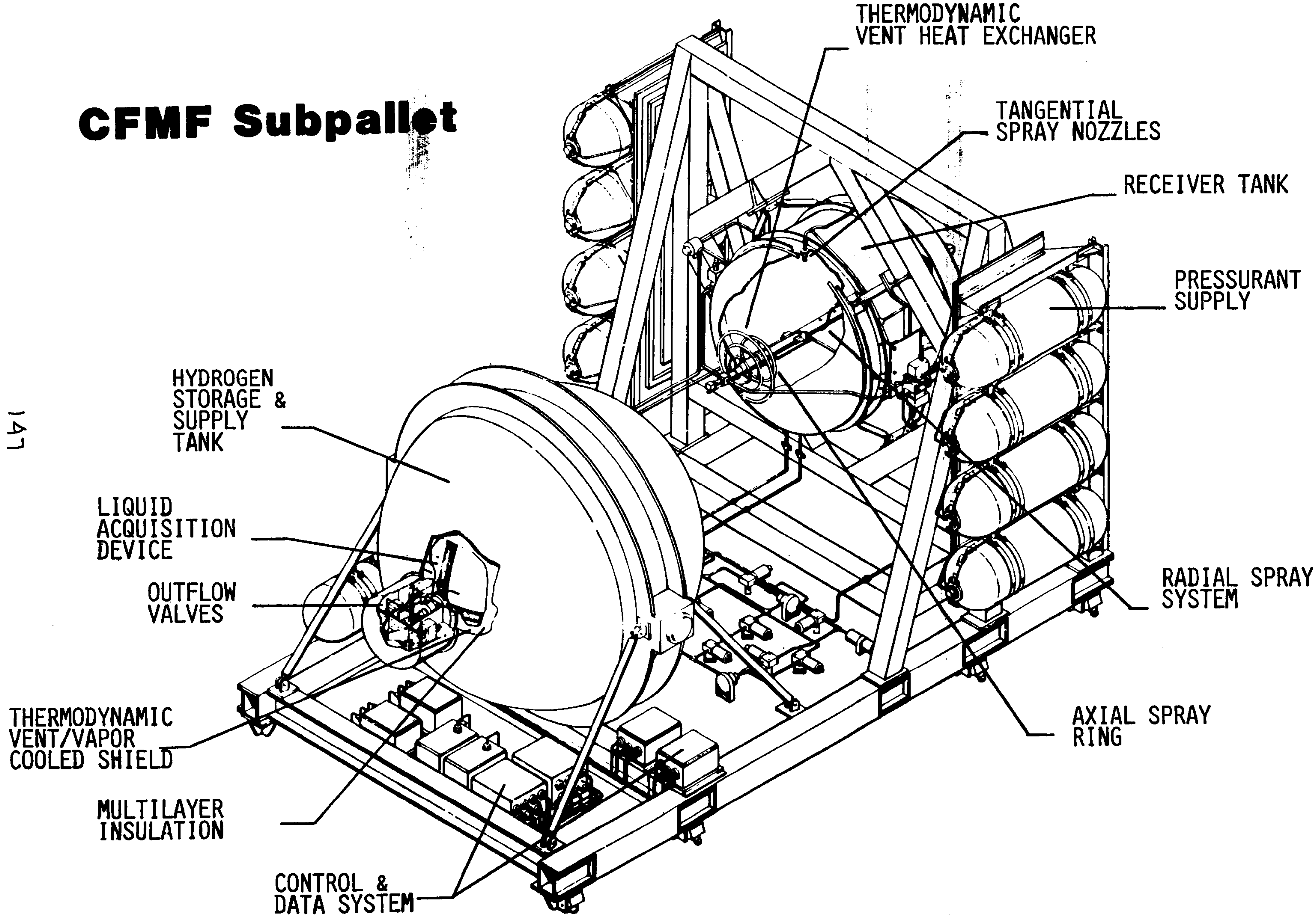
CRYOGENIC FLUID MANAGEMENT FACILITY

APPROACH:

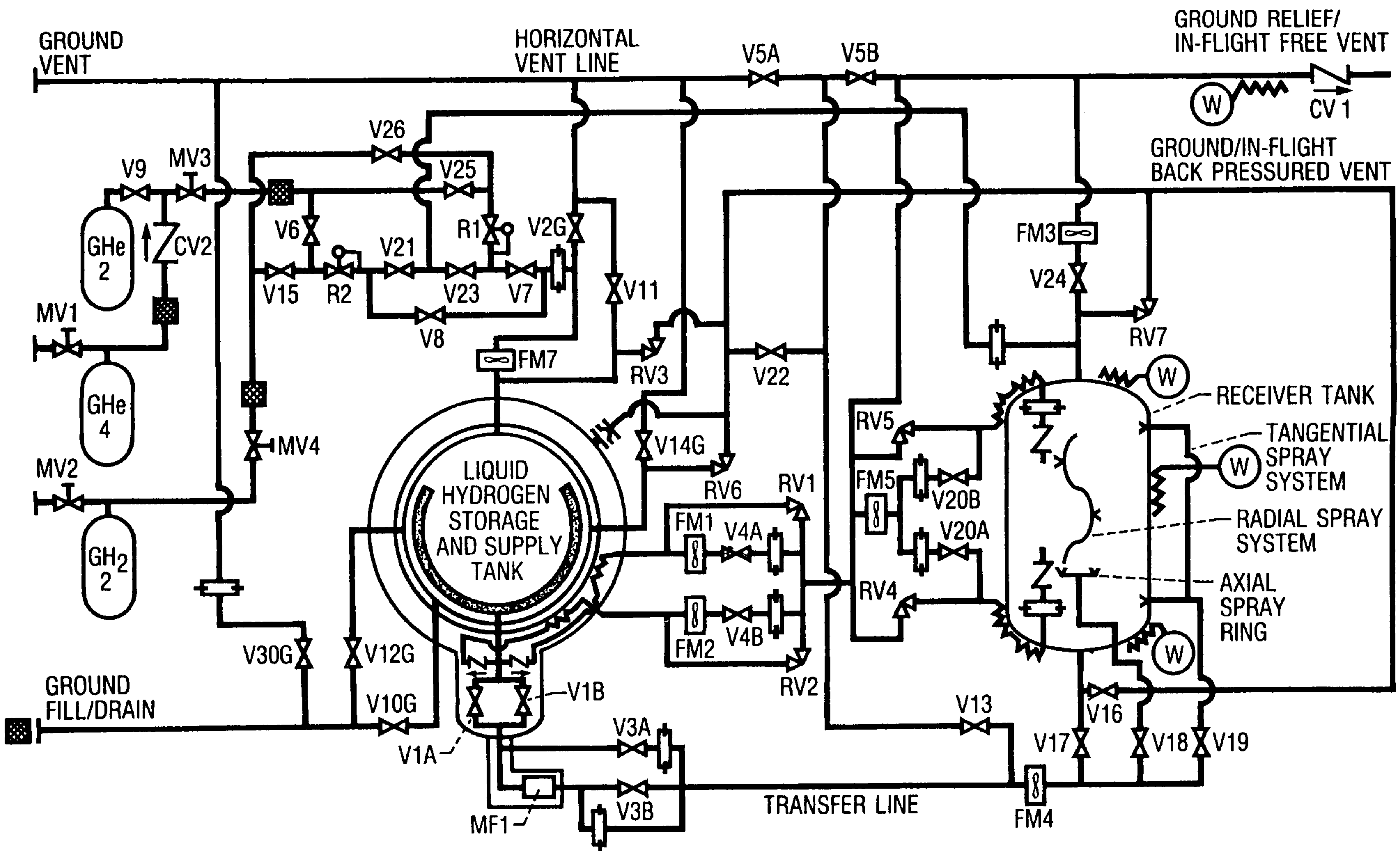
DESIGN, FABRICATE, AND CARRY INTO SPACE A REUSABLE TEST BED TO PROVIDE THE TECHNOLOGY REQUIRED TO MANAGE CRYOGENS IN SPACE.

- CONDUCT EXPERIMENTS IN SPACE TO VERIFY LOW-G FLUID & THERMAL ANALYTICAL MODELS
- VERIFY MODELS TO ESTABLISH DESIGN CRITERIA FOR SUBCRITICAL CRYOGENIC SYSTEMS IN SPACE
- LIQUID HYDROGEN TEST FLUID
- DESIGN FOR SEVEN SHUTTLE FLIGHTS (CURRENT MISSION PLANNING FOR THREE FLIGHTS)
- UTILIZE AVAILABLE EXPERTISE AT LeRC, MSFC, JSC, GSFC, KSC, ARC, JPL

CFMF Subpallet

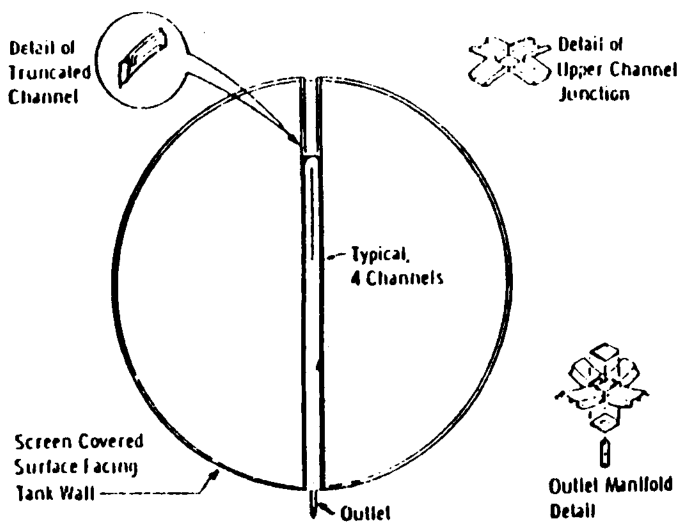


CFMF SIMPLIFIED SCHEMATIC



CRYOGENIC FLUID MANAGEMENT FACILITY

SUPPLY TANK LAD



OBJECTIVES:

- UTILIZE THE DOMINANCE OF SURFACE TENSION FORCES IN A LOW-G ENVIRONMENT TO PROVIDE CONTINUOUS CONTACT (OR COMMUNICATION) WITH THE BULK LIQUID
- PROVIDE SINGLE-PHASE LIQUID OUTFLOW

KEY PARAMETERS:

- SCREEN BUBBLE POINT
- LIQUID RESIDUALS
- SCREEN FLOW AREA
- OUTFLOW RATE
- SYSTEM PRESSURE DROP
- TANK TEMPERATURE AND PRESSURE

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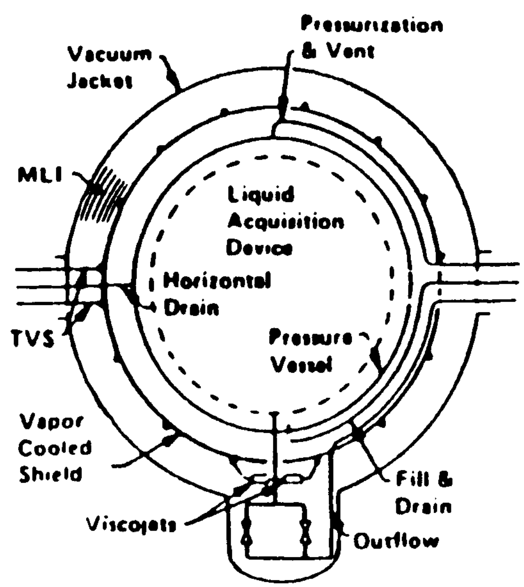
CRYOGENIC FLUID MANAGEMENT FACILITY

THERMAL/PRESSURE CONTROL SYSTEM



OBJECTIVE:

- PROVIDE VAPOR-ONLY VENTING WITHOUT A SETTLING THRUST TO PROVIDE TANK PRESSURE CONTROL IN LOW-G ENVIRONMENT

KEY PARAMETERS:

- JOULE-THOMPSON EFFECT TO INCREASE THE HEAT SINK CAPACITY OF THE VENT FLUID
- FLOW RATE
- TANK TEMPERATURE AND PRESSURE
- HEAT TRANSFER COEFFICIENT OF TWO-PHASE FLUID

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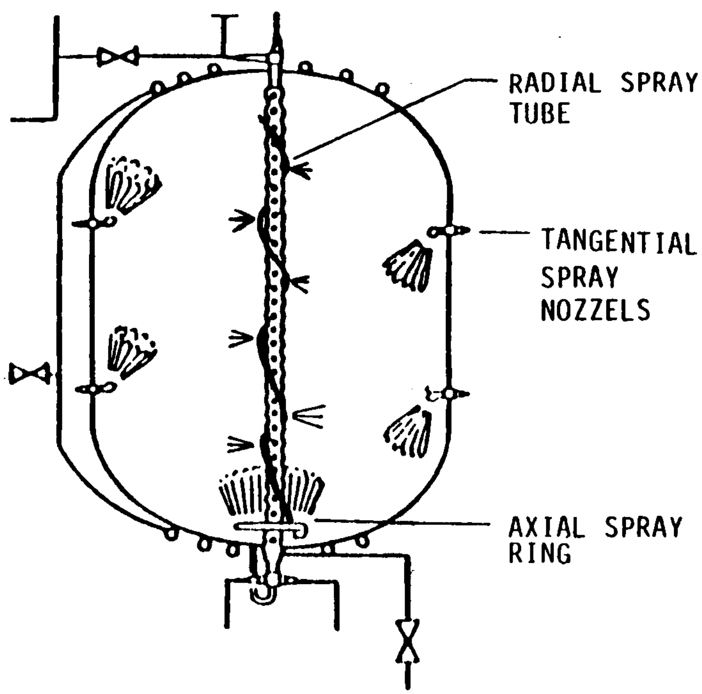
CRYOGENIC FLUID MANAGEMENT FACILITY

RECEIVER TANK CHILLDOWN



OBJECTIVES:

- EFFICIENTLY REDUCE THE TANK STRUCTURE TEMPERATURE TO ALLOW FOR NON-VENTED FILL
- STUDY HEAT TRANSFER RATES AND MECHANISMS IN THE SPACE ENVIRONMENT

KEY PARAMETERS:

- TANK MASS TO VOLUME RATIO
- TANK TEMPERATURE
- LH2 INJECTION TECHNIQUE
- LH2 CHARGE DENSITY

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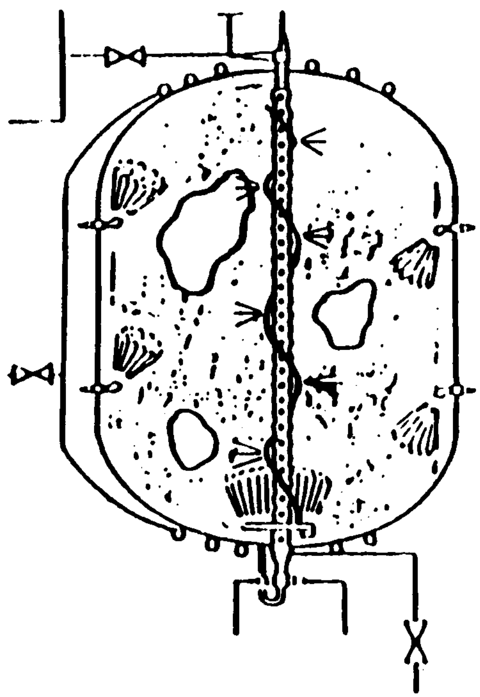
NO-VENT FILL



OBJECTIVES:

- DEMONSTRATE CAPABILITY FOR EFFECTIVE NO-VENT FILL IN LOW-G BY CONDENSATION OF ULLAGE VAPOR

KEY PARAMETERS:

- INITIAL TANK TEMPERATURE
- FINAL TANK PRESSURE
- LH2 INJECTION TECHNIQUE
- L/V INTERFACE AREA
- CONDENSATION RATES

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SPACE EXPERIMENTS OFFICE

CRYOGENIC FLUID MANAGEMENT FACILITY

ANALYTICAL COMPUTER MODELING

OBJECTIVES:

- DEVELOPMENT OF THE CRYOGENIC SYSTEM ANALYSIS MODEL (CSAM) COMPUTER CODE TO ACCURATELY SIMULATE THE PERFORMANCE OF CRYOGENIC SYSTEMS IN THE SPACE ENVIRONMENT

- FLEXIBLE DATA I/O

- VARIFICATION AND REFINEMENT BY CFMF FLIGHT EXPERIMENTS

KEY ANALYSES:

- TRANSIENT HEAT TRANSFER NETWORK

- INTERNAL TANK AND PIPE THERMODYNAMICS

- HEAT EXCHANGER SIMULATION OF TVS

CRYOGENIC FLUID MANAGEMENT FACILITY

POTENTIAL APPLICATIONS

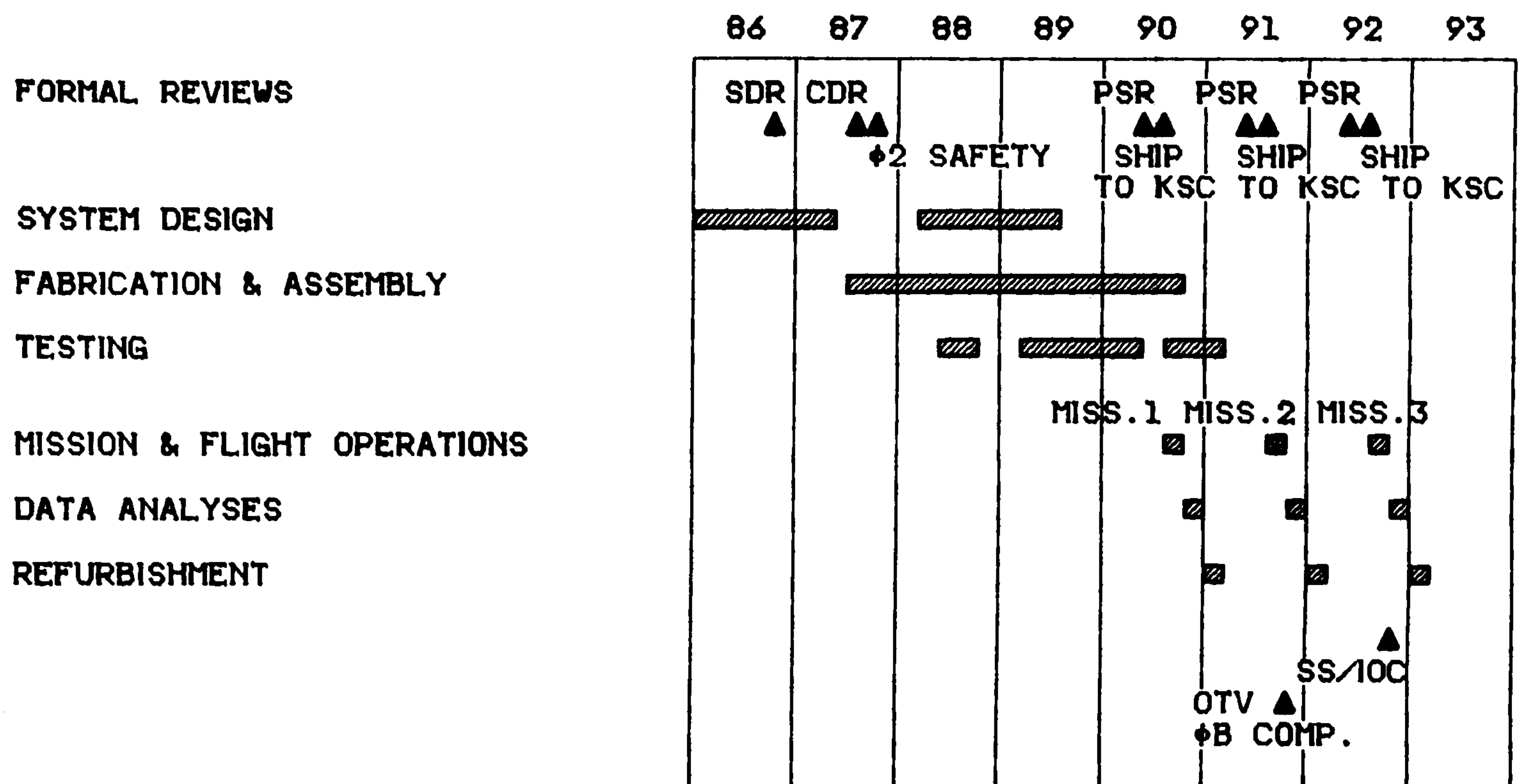
- EARTH-TO-ORBIT TRANSPORT OF CRYOGENIC FLUIDS INCLUDING SCAVENGING (TANKER)
- ORBITAL STORAGE OF CRYOGENIC FLUIDS (DEPOT)
- ON-ORBIT FUELING/REFUELING OF PROPULSIVE STAGES
  SPACE BASED OTV, OMV, ETC./TOP-OFF OF GROUND-BASED VEHICLES
- SPACE STATION SUBSYSTEM CRYOGENIC FLUID REPLENISHMENT
  AUX. PROPULSION/ENERGY STORAGE/LIFE SUPPORT/THERMAL CONTROL
- EXPERIMENT AND SATELLITE CRYOGENIC FLUID SUPPLY AND RESUPPLY
  REACTANTS/COOLANTS/PROPELLANTS
- SHUTTLE ENHANCEMENT
  ORBITAL STAY TIME (POWER AND LIFE SUPPORT)/OMS PERFORMANCE
- SPACE BASED LASER CRYOGENIC FLUID SUPPLY AND RESUPPLY
  REACTANTS/COOLANTS/PROPELLANTS

SPACE EXPERIMENTS OFFICE

NASA

CRYOGENIC FLUID MANAGEMENT FACILITY

MASTER SCHEDULE
(3 MISSIONS)



- SPACE STATION IOC 1992 - GROWTH STATION MID TO LATE 90'S
- OTV IOC PROJECTED FOR 1997 --- PROJECTED PHASE B COMPLETION 1991

ON ORBIT SATELLITE SERVICING

NEAR TERM EQUIPMENT REQUIREMENTS

PRESENTED BY

ADVANCED PLANNING AND ANALYSIS DIVISION
SPACE SCIENCES DEPARTMENT
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION
SCHAUMBURG, IL 60195

AT

SATELLITE SERVICES WORKSHOP II
NASA JOHNSON SPACE CENTER
HOUSTON, TX 77058

NOVEMBER 7, 1985

ACRONYMS AND ABBREVIATIONS

AXAF	ADVANCED X-RAY ASTROPHYSICS FACILITY
BAPS	BERTHING AND POSITIONING SYSTEM
DMSP	DEFENSE METEOROLOGICAL SATELLITE PROGRAM
EMU	EXTRAVEHICULAR MOBILITY UNIT
ERBS	EARTH RADIATION BUDGET SATELLITE
ERS	EARTH RESOURCES SATELLITE
EVA	EXTRAVEHICULAR ACTIVITY
FSS	FLIGHT SUPPORT SYSTEM
GAS	GETAWAY SPECIAL
GEO	GEOSYNCHRONOUS EARTH ORBIT
GRM	GEOPOTENTIAL RESEARCH MISSION
GRO	GAMMA RAY OBSERVATORY
HPA	HOLDING AND POSITIONING AID
HST	HUBBLE SPACE TELESCOPE
IOC	INITIAL OPERATIONAL CAPABILITY
JEA	JOINT ENDEAVOR AGREEMENT
JSC	JOHNSON SPACE CENTER
KSC	KENNEDY SPACE CENTER
LCF	LEASECRAFT/FAIRCHILD
LEO	LOW EARTH ORBIT
LOS	LAND OBSERVATION SATELLITE
MMU	MANNED MANEUVERING UNIT
MPESS	MISSION PECULIAR EQUIPMENT SUPPORT STRUCTURE
MPS	MATERIALS PROCESSING SATELLITE
NASA	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
NOAA	NATIONAL OCEANOGRAPHIC AND ATMOSPHERIC ADMIN.
OMV	ORBITAL MANEUVERING SYSTEM
ORS	ORBITAL REFUELING SYSTEM
OTV	ORBIT TRANSFER VEHICLE
PIP	PAYLOAD INTERFACE PANEL
RMS	REMOTE MANIPULATOR SYSTEM
RSSS	RECONFIGURABLE SATELLITE SERVICING SYSTEM
SIRTF	SPACE INFRARED TELESCOPE FACILITY
SMM	SOLAR MAXIMUM MISSION
SPOT	SYSTEM PROBATOIRE d'OBSERVATION de la TERRE
ST	(HUBBLE) SPACE TELESCOPE
STS	SPACE TRANSPORTATION SYSTEM
TPAD	TRUNNION PIN ATTACHMENT DEVICE
UARS	UPPER ATMOSPHERE RESEARCH SATELLITE

STUDY SYNOPSIS

In April of 1984 the Solar Maximum Mission spacecraft became the first orbiting vehicle to be visited by the Space Shuttle for the purpose of on-orbit servicing. Since that time the errant Westar and Palapa have been recovered and an attempt has been made to activate the dormant Leasat vehicle all from the Space Shuttle. Several experiments and demonstrations have also been carried out on board the Shuttle for other servicing activities such as liquid propellant transfer. And plans are continuing for the on-orbit maintenance of major NASA facilities such as the Hubble Space Telescope and the Gamma Ray Observatory.

The rapidity with which the Space Shuttle has carried out these and other missions has lead to a growing realization among a diverse group of institutions, both governmental and private, both domestic and foreign, that near Earth space has indeed become routinely accessible. In addition these demonstrations by the Space Shuttle of rendezvous, maintenance, repair and/or retrieval have also convinced these institutions that assets of considerable value can be safely located and maintained in orbit. The questions being raised, then, are not "whether" but "how" and "by whom" will these assets be serviced.

The purpose of this study is to determine the potential for servicing all of these diverse assets from the Space Shuttle Orbiter and to assess NASA's role as the catalyst in bringing about routine on-orbit servicing. Specifically this study seeks to determine what requirements, in terms of both funds and time, are needed to make the Shuttle Orbiter not only a transporter of spacecraft but a servicing vehicle for those spacecraft as well.

The scope of this effort is to focus on the near term development of a generic servicing capability. To make this capability truly generic and attractive requires that the customer's point of view be taken and transformed into a widely usable set of hardware. And to maintain a near term advent of this capability requires that a minimal reliance be made on advanced technology.

With this background and scope, this study will proceed through three general phases to arrive at the desired program costs and schedule. The first step will be to determine the servicing requirements of the user community. This will provide the basis for the second phase which is to develop hardware concepts to meet these needs. Finally, a cost estimate will be made for each of the new hardware concepts and a phased hardware development plan will be established for the acquisition of these items based on the inputs obtained from the user community.

STUDY SYNOPSIS

STUDY OBJECTIVE

- TO DETERMINE THE POTENTIAL FOR SATELLITE SERVICING FROM THE SHUTTLE ORBITER FOR ALL USERS AND ASSESS NASA's ROLE AS THE CATALYST TO OPERATIONAL SERVICING

SCOPE OF EFFORT

- FOCUS ON NEAR TERM SERVICING CAPABILITY
- REPRESENT USERS' POINT OF VIEW
- MINIMIZE RELIANCE ON ADVANCED TECHNOLOGY WHICH COULD DELAY APPLICATION

STUDY APPROACH

- DETERMINE USER COMMUNITY REQUIREMENTS
- DEVELOP NEW HARDWARE CONCEPTS WHERE NECESSARY TO ASSURE COMPLETE USER ACCOMMODATION
- FORMULATE A PHASED HARDWARE DEVELOPMENT PLAN WITH COST ESTIMATES

STUDY CONSTRAINTS

In keeping with the objective and scope of the study, several constraints were established at the outset of this effort. The first of these is to focus on the potential servicing missions in the 1986 through 1993 (i.e. Space Station IOC) time frame. Secondly, the candidate servicing missions and hardware concepts should not rely on developments in other programs. This precluded any assistance in the servicing mission from the Orbital Maneuvering Vehicle (OMV), an orbit transfer vehicle (OTV), or the Space Station. This in effect determines a minimum capability needed for on-orbit servicing and sets the stage for expanded capability when these other systems become available.

STUDY CONSTRAINTS

- FOCUS EFFORT ON SERVICING WHICH CAN BE DONE FROM STS IN NEAR TERM
  - 1986 THROUGH 1993 TIME FRAME
- DO NOT RELY ON DEVELOPMENTS IN OTHER PROGRAMS
  - NO OMV, OTV, OR SPACE STATION
  - DETERMINE MINIMUM CAPABILITY NEEDED

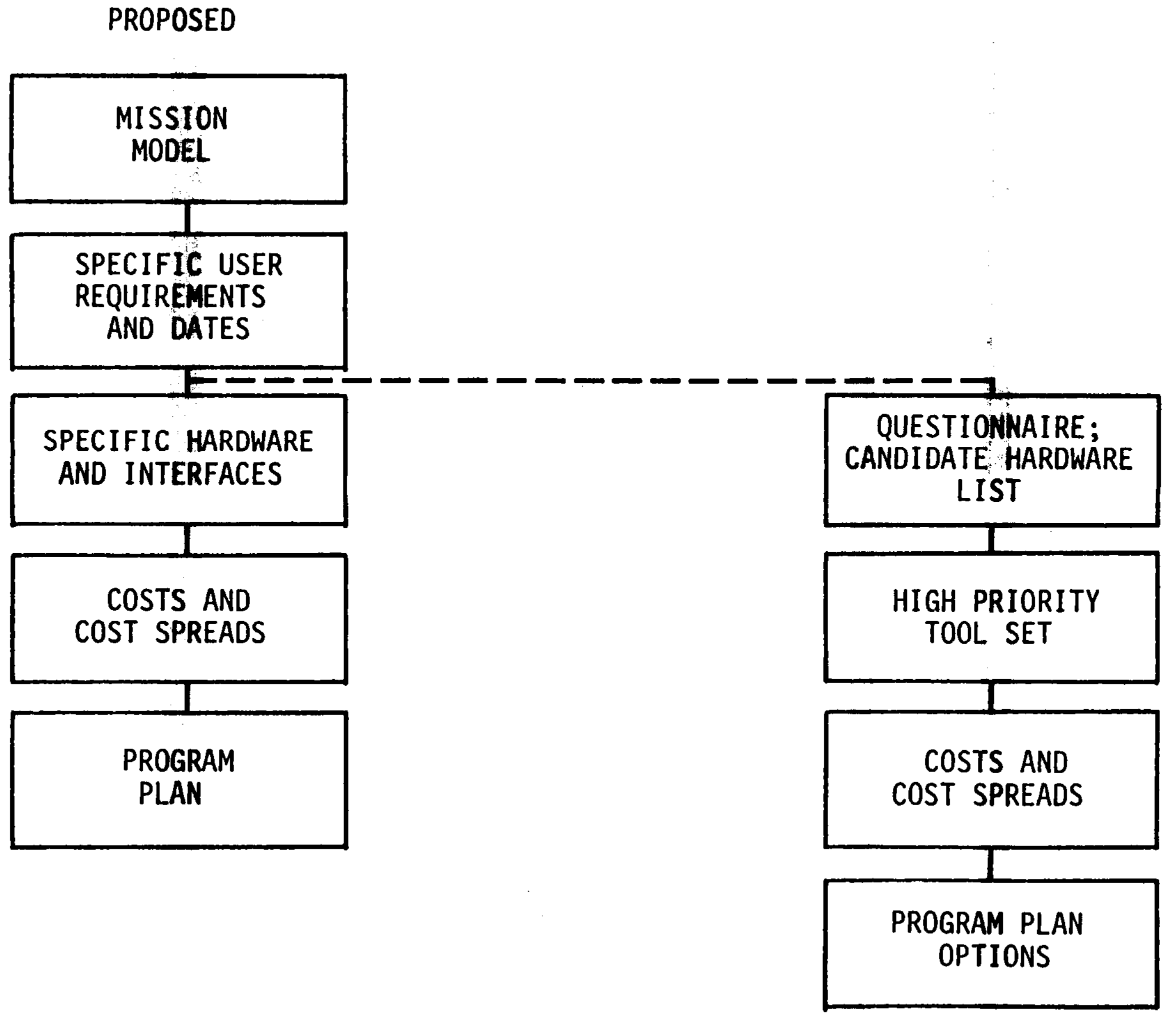
STUDY PROCESS

As originally proposed, this study was divided into five specific tasks each with a well defined end product. The first of these tasks would ascertain what missions would be flown in the time period of interest. These missions would be divided into those which would definitely require servicing, those which would be serviced on a contingency basis or could be designed to take advantage of servicing, and finally those which, for whatever reason, were not servicable. The second task would take the first two groups of missions and, based on published information or contact with the user, determine specific user requirements and servicing dates. The third task would match hardware to these user requirements and identify all interfaces between the user and the servicing equipment. The fourth task would estimate the cost and three year cost spreads for any new hardware items required to meet user needs. Finally, the servicing dates provided by the users would be utilized to develop a satellite servicing program plan for the development and acquisition of all hardware elements.

When put into action, the first and second tasks were implemented as planned. However, it was determined in the second task that, with the exception of a few missions, specific servicing requirements were not known. Many potential customers were still analyzing their options and assumed that they would adapt to hardware already in existance. These potential customers were generally reluctant to level specific servicing requirements on NASA due to uncertainty in the operational date for that hardware and potential development charges assessed by NASA on the first customer.

To circumvent this situation and still obtain user community inputs, a "what if" scenario was devised and a questionnaire prepared for distribution to the potential users. This questionnaire, the actual third task in this study, combined a list of existing and new hardware concepts which the users were asked to rate for acceptability to their servicing needs. Results from this questionnaire will be presented in subsequent sections of this report. These results allowed a high priority tool set to be drawn from the tool list (actual Task 4) and costed as originally planned. Since no firm requirements were generated as part of this questionnaire, the last task prepared various program plan options based on assumptions generated from the questionnaire.

STUDY PROCESS



QUESTIONNAIRE RECIPIENTS

The facing page indicates the organizations with which the questionnaire recipients are associated. Below each organization name is a list of the type of LEO spacecraft with which that organization has had experience.

QUESTIONNAIRE RECIPIENTS

BALL AEROSPACE

- NASA
- COMMERCIAL

FAIRCHILD SPACE COMPANY

- NASA
- COMMERCIAL

GRUMMAN AEROSPACE

- NASA

LOCKHEED MISSILES AND SPACE CO.

- NASA
- DoD

MARTIN-MARIETTA CORP.

- NASA
- DoD

NASA/JSC ASTRONAUT OFFICE

RCA ASTROELECTRONICS

- COMMERCIAL
- NOAA
- DoD

ROCKWELL INTERNATIONAL

- NASA

SPAR AEROSPACE

- NASA
- ESA

TRW FEDERAL SYSTEMS DIVISION

- NASA

USAF SPACE COMMAND

- DoD

USAF SPACE DIVISION

- DoD

KEY FINDINGS AND RESULTS

Before presenting any details of the study tasks, a set of key findings and results from all of the tasks will be discussed to both help introduce and explain the direction taken by each task.

Between 1986 and 1993, 63 missions were identified from NASA, NOAA, DoD, U.S. commerical and foreign institutions which could be reached by, or could potentially fly down to, the Space Shuttle Orbiter. Thirty-three of these missions were identified as potentially serviceable: 12 on a regular or scheduled basis, 21 on a contingency basis. If all of these missions are flown at their presently scheduled times, there appears to be sufficient traffic for up to 20 servicing sorties per year in 1993. Since many of these vehicles are in the early design phase, precise servicing requirements and interfaces are not known for these vehicles. As servicing hardware and procedures are developed for other missions, the design of these vehicles will evolve to take advantage of these developments. However, many potential customers remain undecided regarding the degree to which their spacecraft should be serviced on-orbit.

KEY FINDINGS AND RESULTS

- 33 VEHICLES POTENTIALLY SERVICEABLE
  - 12 ON A REGULAR BASIS
  - 21 ON A CONTINGENCY BASIS
- APPEARS TO BE SUFFICIENT TRAFFIC TO SUPPORT UP TO 20 SERVICING SORTIES PER YEAR
- PRECISE SERVICING REQUIREMENTS AND INTERFACES NOT KNOWN IN MAJORITY OF CASES
- MANY POTENTIAL CUSTOMERS REMAIN UNDECIDED REGARDING ON-ORBIT SERVICING

KEY FINDINGS AND RESULTS (continued)

The reason for this indecision can be traced back to several key features of NASA policy. First, there is no NASA policy statement in existance regarding the routine (operational) on-orbit servicing for non-NASA vehicles. The non-NASA community is thus uncertain as to the degree of commitment NASA will make in the long term for the on-orbit servicing of their vehicles. Secondly, the user community has indicated, through the questionnaire mentioned previously, that a pricing policy for on-orbit servicing is of equal or greater importance than specific knowledge of hardware capabilities or interfaces. This pricing policy is critical for the determination of the level of servicing and no such policy currently exists. As long as the formulation of this pricing policy is deferred, customers will try to minimize their uncertainty and risk by opting for non-serviceable or minimally serviceable spacecraft. Finally, most non-NASA users are waiting to see what hardware NASA will design, develop and acquire for servicing. These users will tend not to level requirements on NASA for fear that NASA will burden them with the development cost as the first user. These customers feel thay they have sufficiently clever engineers to make the best use of existing hardware or to design and develop hardware for their specific use.

KEY FINDINGS AND RESULTS (CONTINUED)

- NO NASA POLICY REGARDING ROUTINE (OPERATIONAL) ON-ORBIT SERVICING FOR NON-NASA PAYLOADS EXISTS

- PRICING POLICY IS OF EQUAL OR GREATER IMPORTANCE THAN PRECISE KNOWLEDGE OF HARDWARE INTERFACES AND CAPABILITIES

- AS LONG AS PRICING POLICY IS DEFERRED, CUSTOMERS WILL TEND TO OPT FOR NON-SERVICEABLE OR MINIMALLY SERVICEABLE SPACECRAFT

- NON-NASA USERS WAITING TO SEE WHAT NASA WILL DESIGN, DEVELOP AND ACQUIRE AS GENERIC HARDWARE
    - USERS WILL NOT LEVEL REQUIREMENTS ON NASA
    - USERS WILL REACT TO HARDWARE ALREADY AVAILABLE

KEY FINDINGS AND RESULTS (continued)

The user community thus feels that the next move is up to NASA to break the current "Catch-22" situation (NASA waiting for customers to specify requirements so that hardware can be built; customers waiting for hardware and pricing policy so that the most cost effective vehicle can be designed). If the customers perceive that NASA will not take the next step, then they will opt for non-serviceable spacecraft or will design and build their own unique servicing hardware. From NASA's perspective, this latter situation could mean a proliferation of hardware interfaces and servicing procedures as each new vehicle is developed.

KEY FINDINGS AND RESULTS (CONTINUED)

- NEXT MOVE IS NASA'S; THE RIGHT RESPONSE WILL BREAK THE "CATCH-22" SITUATION

- IT IS IMPORTANT TO REMEMBER THAT NASA IS SELLING A SERVICE
  - IF CUSTOMERS ARE DISSATISFIED, THEY WILL DO WITHOUT OR WILL BUILD THEIR OWN HARDWARE
  - IF THE CUSTOMERS BUILD THEIR OWN HARDWARE, THERE IS A RISK OF PROLIFERATING FUNCTIONALLY SIMILAR BUT PHYSICALLY DIFFERENT HARDWARE

- NASA AND CUSTOMERS CAN WORK TOGETHER IF NASA DEFINES THE GROUND RULES (POLICY); EXAMPLES INCLUDE:
  - CUSTOMER DEVELOPED HARDWARE LEASED TO NASA OR RENTED TO OTHER USERS
  - NASA PURCHASED HARDWARE WITH CUSTOMER LEASE-BACK OPTIONS
  - NASA DEVELOPED PROTOTYPES, CUSTOMER PURCHASED FLIGHT ARTICLES

KEY FINDINGS AND RESULTS (continued)

If NASA decides to offer servicing to non-NASA customers as a standard STS feature, then the user community questionnaire indicates that there are several highly desirable attributes and components which should be part of this service. The customers will expect to see identical interfaces at the STS, OMV, OTV and Space Station. A strong preference was also indicated for the following hardware items:

(1) A payload interface panel which will allow more direct and sophisticated communication with the spacecraft being serviced by those on board the Space Shuttle. This will allow the vehicle to be monitored during servicing and checked when all servicing has been completed.

(2) A second arm either in the form of a second RMS (preferred) or an HPA mounted on a cradle. This will allow vehicles to be serviced outside of the payload bay envelope and avoid a potential charge to reserve clearance space within the payload bay.

(3) Remote controlled fluids coupling. This would avoid EVA time to connect and disconnect such a coupling.

(4) A large (5000 lb minimum capacity) monopropellant tanker. Vehicles which potentially need refueling require anywhere from 2000 to 5000 lb of propellant. No missions using bipropellants or cryogens were identified.

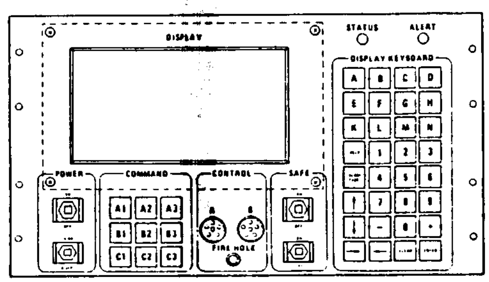
(5) Large spacecraft berthing device. This device was identified for very massive vehicles (30000-40000 lb) which would be berthed to the orbiter for extended periods of time (6-8 hrs or more).

KEY FINDINGS AND RESULTS (CONTINUED)

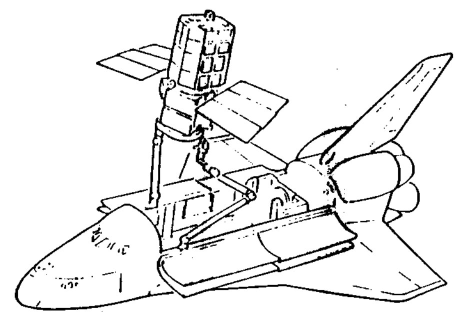
IF NASA DECIDES TO SERVICE NON-NASA PAYLOADS:

- USERS EXPECT TO SEE IDENTICAL INTERFACES AT STS, OMV, AND SPACE STATION
- HIGHLY DESIRABLE HARDWARE DEVELOPMENTS FOR NEAR TERM SERVICING

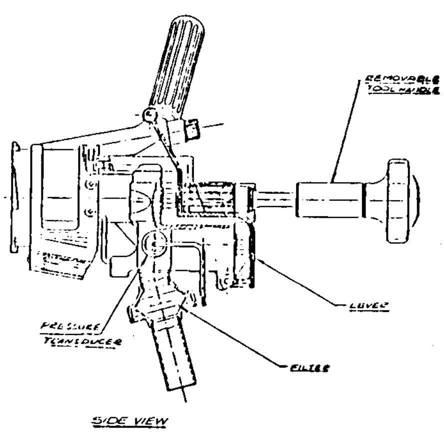
PAYLOAD INTERFACE PANEL



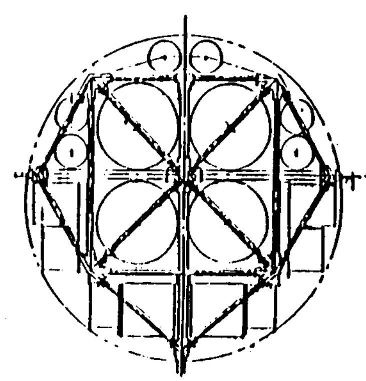
2ND ARM



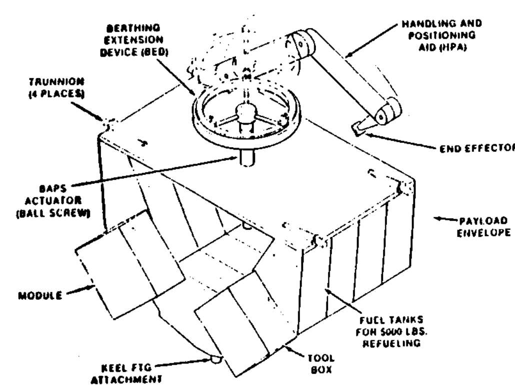
FLUIDS COUPLING



LARGE TANKER



LARGE SPACECRAFT BERTHING DEVICE



KEY FINDINGS AND RESULTS (continued)

The advantages to NASA in taking this approach will be to not only expand the uses for the Space Shuttle but also to acquire practical experience in designing and operating servicing hardware which could be used by the OMV and Space Station. This will also cultivate a sense of confidence in the servicing concept among those customers who will ultimately be based at the Space Station or will be accessible only by means of the OMV or OTV.

KEY FINDINGS AND RESULTS (CONTINUED)

- ADVANTAGES TO NASA
  - ACQUIRE EXPERIENCE IN PROCEDURES AND HARDWARE TO BE USED BY SPACE STATION, OMV AND OTV
  - CULTIVATE USER CONFIDENCE IN STS SERVICING CAPABILITIES WHICH WILL TRANSLATE INTO EARLIER SERVICABLE SPACECRAFT DESIGNS
  - THIS WILL LEAD TO FULLER UTILIZATION OF SERVICING IN THE SPACE STATION, OMV AND OTV OPERATIONAL ENVIRONMENT

KEY FINDINGS AND RESULTS (concluded)

Because of the lack of firm servicing requirements among potential customers at the present time, no single comprehensive program plan could be assembled. Rather, a set of plans were developed which bounded the range of possibilities and also examined the middle ground. At one end of the range is a "NASA-only" scenario in which only those tools which would support NASA programs would be developed (and this excludes tools currently under development for HST). This represents the situation where NASA maintains the status quo regarding non-NASA customers and thus becomes a baseline case against which other programs can be compared (Scenario 1). At the other extreme is a scenario driven by identified needs within the commercial community and assumes that spacecraft will rely on potential servicing capability to the maximum extent possible (Scenario 2, the "Fast-Start Commercialization" option). In this scenario seven of the nine identified hardware items would be operational by the end of Fiscal Year 1989 (FY89). This drives the peak year to FY88 and consumes 40 percent of the total cost during that year. A possible compromise (Scenario 3) would retain the same ultimate capability as the Fast-Start case but would stretch out the development of various hardware items to be more in line with the current NASA schedule. These three scenarios are compared on the facing page and more information concerning each plan will be presented later in this report.

KEY FINDINGS AND RESULTS (CONCLUDED)

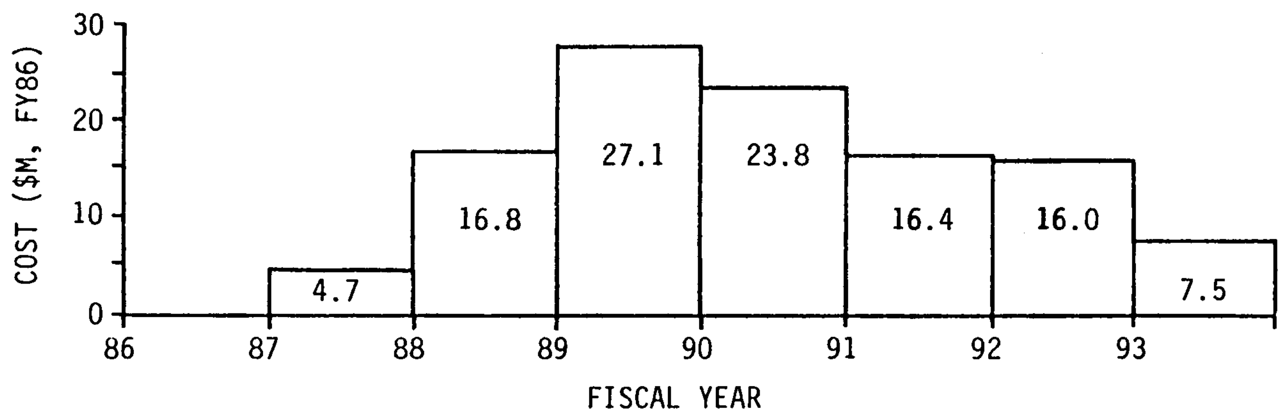
PROGRAM PLAN OPTIONS

- RANGE OF OPTIONS VARIES FROM NASA-ONLY PROGRAM TO FAST-START COMMERCIALIZATION

	NUMBER OF TOOLS	TOTAL COST (THRU FY93)	PEAK COST	PEAK YEAR
NASA-ONLY*	6	65 M (FY86)	20 M (FY86)	FY89
FAST-START*	9	125 M (FY86)	51 M (FY86)	FY88

* ASSUMES TWO FLIGHT ARTICLES

- POSSIBLE COMPROMISE - SAME ULTIMATE CAPABILITY AS FAST-START BUT SOME ITEMS DEFERRED TO LATER YEARS (ASSUMES TWO FLIGHT ARTICLES)



STUDY RECOMMENDATIONS

Based on the information gathered from potential customers and data generated as part of this study, the following course of action is recommended.

(1) NASA should formally commit to routine on-orbit servicing of non-NASA spacecraft as a standard feature of the STS. Potential customers already expect NASA to do this, but are waiting for a formal announcement. As part of the this commitment, a focal point within NASA must be identified to be cognizant of servicing related activities within NASA and to serve as a point of contract for non-NASA customers.

(2) A set of generic servicing equipment should be baselined with their capabilities made known to the potential customers. These customers will not design their vehicles for a particular type of servicing if there is a risk that the necessary equipment will not be available at the appropriate time. The user community survey indicated that five hardware items are of particular interest to this community. These items include a payload interface panel, a second arm, a monopropellant tanker with at least 5000 lb. capacity, a remote controlled fluids coupling, and a heavy berthing/docking fixture. This equipment should be available from all Shuttle Orbiters to avoid manifesting limitations. As servicing expands beyond the Shuttle Orbiter, equipment features should be the same wherever possible for servicing done by an OMV, OTV or the Space Station. Customers will then not be forced to select the servicing vehicle when their spacecraft is designed.

STUDY RECOMMENDATIONS

RECOMMENDED COURSE OF ACTION BASED ON CUSTOMERS' INPUTS

(1) COMMIT NASA TO ROUTINE SERVICING FOR NON-NASA SPACECRAFT

- IDENTIFY AGENCY FOCAL POINT FOR SERVICING RESPONSIBILITY

(2) BASELINE GENERIC SERVICING EQUIPMENT

- CUSTOMER WILL NOT DESIGN FOR SERVICING AT RISK OF NOT HAVING EQUIPMENT AVAILABLE

- PRIORITIZE AND COMMIT TO DEVELOPMENT OF HIGHLY DESIRABLE HARDWARE INCLUDING: TANKER, 2ND RMS OR HPA, HEAVY BERTHING AND DOCKING FIXTURE, REMOTE CONTROLLED FLUIDS UMBILICAL, AND PAYLOAD INTERFACE PANEL

- EQUIPMENT FEATURES SHOULD BE THE SAME (IF POSSIBLE) FOR SERVICING HARDWARE USED BY THE STS, SPACE STATION, OMV AND OTV

STUDY RECOMMENDATIONS (continued)

(3) With a set of baselined servicing equipment in place, the potential customers must be given a set of cost guidelines for the use of this equipment and for any other non-hardware related items which must be paid for (i.e. transportation to orbit, training, etc). As indicated by the questionnaire, the user community places equal or greater value on this information that on hardware details since cost trades must be performed early in the design cycle.

(4) One of the results from the questionnaire was the perception by the user community that NASA would lead both hardware development and on-orbit operations. The DoD is also assuming that a hardware development program is underway within NASA and that this hardware will be available when the DoD decides the type and volume of servicing its wishes to carry out. These two perceptions indicate that a dialog between NASA and the non-NASA community must be opened to avoid misconceptions and to allow for appropriate planning by all involved. The needs and roles of all organizations who will eventually be involved with on-orbit servicing are currently in a dynamic state and will remain so for the next several years. A regularly scheduled forum where needs, desires and constraints can be aired will help each organization to define its own options and may allow for cooperative endeavors to be arranged.

(5) Finally, interfaces for servicing equipment must be standardized. The customer will need this kind of information to properly design their spacecraft for compatibility. Standardization will allow the customer to use the same automated or remotely operated equipment regardless of orbit location (LEO, polar, GEO, etc.) or servicing vehicle (Shuttle Orbiter, OMV, OTV, or Space Station). The details of these interface standards is one of the tasks which could be worked out by the government/industry forums suggested in item (4).

STUDY RECOMMENDATIONS (CONCLUDED)

(3) ESTABLISH SERVICING COST OR PRICING GROUND RULES

- CUSTOMERS REQUIRE THIS INFORMATION NOW FOR TRADE STUDIES

(4) BEGIN GOVERNMENT/INDUSTRY SERVICING MEETINGS

- NEEDS AND ROLES OF VARIOUS ORGANIZATIONS ARE IN A DYNAMIC STATE AND WILL REMAIN SO FOR NEXT SEVERAL YEARS
- MAINTAIN FLOW OF INFORMATION REGARDING DESIRES, REQUIREMENTS AND CONSTRAINTS

(5) STANDARDIZE SERVICING INTERFACES

- CUSTOMERS NEED GUIDELINES TO DESIGN FOR SERVICING AND BE COMPATIBLE WITH SERVICING EQUIPMENT
- AUTOMATED OR REMOTELY OPERATED INTERFACES SHOULD BE SAME FOR ALL SERVICEABLE SPACECRAFT: LEO, POLAR, GEO, ETC.
- DETAILS CAN BE WORKED OUT IN GOVERNMENT/INDUSTRY SERVICING MEETINGS TO SATISFY LARGEST POSSIBLE USER GROUP

ON ORBIT EXCHANGE OF LARGE MODULES

THE THREE BODY PROBLEM

PRESENTED BY

ADVANCED PLANNING AND ANALYSIS DIVISION
SPACE SCIENCES DEPARTMENT
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION
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AT

SATELLITE SERVICES WORKSHOP II
NASA JOHNSON SPACE CENTER
HOUSTON, TX 77058

NOVEMBER 7, 1985

ACRONYMS AND ABBREVIATIONS

EVA	EXTRA-VEHICULAR ACTIVITY
FSC	FAIRCHILD SPACE COMPANY
FSS	FLIGHT SUPPORT SYSTEM
HPA	HANDLING AND POSITIONING AID
ORU	ORBITAL REPLACEMENT UNIT
PBS	PAYLOAD BERTHING SYSTEM
RCA	RCA ASTROELECTRONICS DIVISION
RMS	REMOTE MANIPULATOR SYSTEM
RSSS	RECONFIGURABLE SATELLITE SERIVCING SYSTEM
STS	SPACE TRANSPORTATION SYSTEM

INTRODUCTION

The three body problem as used in this study refers to the on-orbit exchange of very large payload modules from a free flying support bus. Thus, the three bodies include the large on-orbit payload, the exchange payload and the support bus. Each of these modules must be manipulated in one way or another by the shuttle orbiter and its crew.

The payload modules have been defined here as having a length of approximately 30 ft and/or a weight in the 20,000 to 30,000 lb range. The free flying support bus would have a weight of 10,000 to 20,000 lb depending upon the propellant load and would provide on-orbit services such as electrical power, environmental control, pointing, and communications.

The need for a solution to this problem arose from two different sources. The near-term application of such a solution would be applied to commercial materials processing payloads. With a commercial payload of this size, the desire to limit costs associated with the payload exchange has become a driving factor. A long-term application of this concept will be to support the Space Station unmanned platform. The polar platform will obviously require the shuttle orbiter to conduct any exchange with incumbent platform payloads. The co-orbiting platform, on the other hand, will generally be serviced at the Space Station. However, some of the larger NASA payloads and future commercial microgravity factories may find it more suitable to conduct such an exchange at the Shuttle orbiter rather than go through the intermediate step of transferring the payload at the Space Station.

INTRODUCTION

- 3 BODIES: LARGE, ON-ORBIT MODULE, REPLACEMENT MODULE, SUPPORT BUS

- LARGE MODULE: 20,000 TO 30,000 LB WEIGHT
  UP TO 30 FEET IN LENGTH

- SUPPORT BUS: 10,000 TO 20,000 LB WEIGHT

- SOLUTION NEEDED BY:

  NEAR-TERM - MATERIALS PROCESSING FACTORIES

  FAR-TERM - SPACE STATION UNMANNED PLATFORM

TASK OBJECTIVE AND SCOPE

Three major objectives have been set for this task. The first of these is to gather servicing requirements from potential users. This set of requirements can thus serve as a starting point for any future studies in this area. These requirements will also serve as a starting point for this task, indicating the type of hardware and exchange procedures which are viable from the user's perspective.

The second objective is to take the information from the first objective, along with the general description of the mission to be accomplished, and formulate possible scenarios to conduct the module exchange. Each of these scenarios will require certain hardware items to complete the task. Part of this objective, then, is to identify generic hardware concepts which could meet the exchange scenario requirements. Both the scenario and hardware options should cover the range of possibilities in a general sense. Any fine tuning of a particular concept or procedure can be made in future efforts.

The final objective of this task is to gather and summarize the reactions to both the hardware and procedures from potential users. This will serve to narrow down the range of possible options for any future study.

The scope of this task covers only the exchange of large payload modules. Other types of servicing, such as subsystem module replacement or support bus refueling, which may occur as part of the module exchance mission have not been addressed. In addition, hardware capabilities will be derived primarily from near-term user requirements. Other far-term users will presumable adapt to, or make minor adjustments in, these capabilities.

TASK OBJECTIVES AND SCOPE

- OBJECTIVE
  - GATHER POTENTIAL USER INPUT REGARDING SERVICING REQUIREMENTS
  - FORMULATE EXCHANGE SCENARIOS AND NECESSARY HARDWARE
  - SUMMARIZE USER PREFERENCES REGARDING SCENARIOS AND HARDWARE
- SCOPE
  - EXAMINE HARDWARE AND PROCEDURE OPTIONS FOR MODULE EXCHANGE ONLY
  - FOCUS ON REQUIREMENTS FROM NEAR-TERM USERS

POTENTIAL VEHICLE AND MISSION CANDIDATES

At the present time three vehicles have been identified as possible users for this type of large module exchange. Two of these are commercially operated spacecraft whose primary customers will probably be materials processing payloads. The third vehicle is the NASA unmanned platform being developed in conjunction with the Space Station. Most of the currently identified users of this platform are large NASA science payloads.

Both Fairchild Space Company and RCA AstroElectronics have proposed spacecraft which would lease on-orbit services (i.e. electrical power, environmental control, communications, etc.) to various payloads. The vehicle by itself would weigh in the range of 10,000 lb to 20,000 lb depending upon the propellant load. A typical payload for this type of spacecraft would be a continuous flow electrophoresis factory being developed by McDonnell Douglas Astronautics Corp. and the Ortho Division of Johnson & Johnson. This factory would weigh approximately 25,000 lb and would require a payload exchange every three to six months.

The third vehicle in this category is the NASA unmanned platform. NASA's reference configuration for this spacecraft has an estimated dry weight of 12,000 lb. Accommodations have been made for up to 6,000 lb of propellant which would be used basically to raise and lower the vehicle's orbit altitude. This yields a total platform weight of 18,000 lb without a payload. The larger payloads identified for this vehicle thus far are NASA science observatories. These include the Advanced Solar Observatory (20,000 lb, 28.5 deg inclination), the High Throughput Experiment (22,000 lb, 28.5 deg inclination) and the Polar C collection of instruments (10,000 lb, 98 deg inclination).

POTENTIAL VEHICLE/MISSION CANDIDATES

- LEASECRAFT (FSC)
  - SUPPORT BUS: APPROXIMATELY 20,000 LB TOTAL WEIGHT; 5,000 LB PROPELLANT
  - TYPICAL PAYLOAD: MATERIALS PROCESSING FACTORY; APPROXIMATELY 25,000 LB TOTAL WEIGHT
- OMNISTAR (RCA)
  - COMPARABLE TO LEASECRAFT IN SIZE AND PAYLOAD
- SPACE STATION UNMANNED PLATFORM
  - NASA REFERENCE CONFIGURATION: 12,000 LB DRY WEIGHT; UP TO 6,000 LB PROPELLANT
  - BOTH HIGH AND LOW INCLINATION ORBIT LOCATIONS
  - POTENTIAL LARGE PAYLOAD: ADVANCED SOLAR OBSERVATORY, 20,000 LB

INITIAL USER REQUIREMENTS FOR THE THREE-BODY PROBLEM

In addition to the already available RMS arm, a berthing/docking interface for the platform has been indicated as a minimum requirement by potential users, preferably outside of the revenue-generating Shuttle payload bay envelope. A temporary storage location for the exchange module is also required. This hardware could be located in the payload bay, or as an additional out-of-bay berthing device. In-bay berthing devices are constrained by requirements for clear access to the modules during grappling, berthing and redeployment operations. Further, attention must be given to visibility considerations for operations using in-bay hardware, and especially for out-of-bay operations which may occur during orbit "nighttime".

All mechanical interfaces must be compatible as well as simple and cheap. Standard trunnion/latch interfaces would be desirable. A communication link and power supply to the platform and modules are required during changeout operations. Also, changeout operations must be completed in approximately 3 hours, as no module should be without power in excess of approximately 1 hour. During a module exchange mission, it may also be necessary or prudent to replenish the platform propulsion system modules, as a capability for fluid exchange may be required. Potential users also indicated that operations should be designed to minimize crew activity which will in turn minimize crew training. The platform/module exchange system should also be designed to take full advantage of the STS nominal launch manifest and attempt to avoid any requirement for unscheduled launches.

The bay requirement for missions of this type is to keep the costs low. Hardware development costs can be lowered by using existing flight support equipment such as the RMS, FSS and Spacelab pallets. Transportation (launch) costs could be minimized by careful optimization of weight load factors, length load factors, crew interaction (minimize EVA's and crew training) and mission unique flight plans. For example, the flight support equipment could be packaged with the new module in such a way that its launch cost is included in the weight and length load factors attributed to the alone.

INITIAL USER REQUIREMENTS FOR THE THREE-BODY PROBLEM

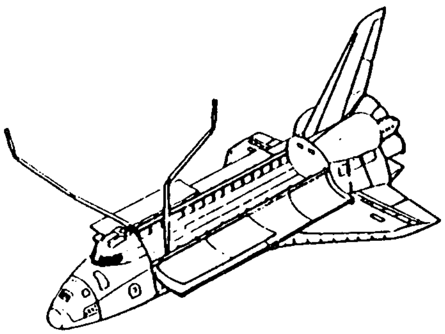
- HARDWARE REQUIREMENTS
  - PROVIDE A BERTHING/DOCKING INTERFACE FOR THE PLATFORM
  - PROVIDE A TEMPORARY STORAGE LOCATION FOR THE EXCHANGE MODULE
- HANDLING, BERTHING AND DOCKING PROVISIONS
  - SIMPLE (AND CHEAP) MECHANICAL INTERFACES
  - ELECTRICAL INTERFACE: PROVIDE POWER TO THE MODULE
PROVIDE COMMAND/MONITORING OF THE MODULE
  - POSSIBLE FLUID EXCHANGE
  - VISIBILITY AND FREE ACCESS CONSIDERATIONS
- OPERATIONAL REQUIREMENTS
  - PERFORM CHANGEOUT IN APPROXIMATELY 3 HOURS
  - TAKE ADVANTAGE OF NOMINAL STS LAUNCH MANIFEST
  - MINIMIZE CREW ACTIVITY
- COSTS
  - LOW LAUNCH COSTS
  - LOW HARDWARE DEVELOPMENT COSTS

SERVICING HARDWARE DESCRIPTIONS

1. REMOTE MANIPULATOR SYSTEM (RMS) -		LENGTH = 50 FT, WEIGHT = 905 LB, TRANS COST = $1.76M* (FOR 2ND RMS)
2. HANDLING AND POSITIONING AID (HPA)		
OPTION A:	RMS-BASED	LENGTH = 22 FT, WEIGHT = 802 LB, TRANS COST = $1.51M
OPTION B:	"STRONG-ARM"	LENGTH = 20 FT, WEIGHT = 1820 LB, TRANS COST = $3.43M
OPTION C & D:	A & B WITH CRADLE	REQUIRED PAYLOAD BAY LENGTH = 4 FT, WEIGHT = 1000 LB, TRANS COST = $9.19M
3. IN-BAY DOCKING FIXTURES		
OPTION A:	FLIGHT SUPPORT SYSTEM (FSS)	REQ'D PAYLOAD BAY LENGTH = 1.5 FT, WEIGHT = 3300 LB, TRANS COST = $6.22M
OPTION B:	LIGHTWEIGHT FSS	REQ'D PAYLOAD BAY LENGTH = 2 FT, WEIGHT = 2000 LB, TRANS COST = $5.1M
4. OUT-OF-BAY DOCKING FIXTURES		
OPTION A:	PAYLOAD BERTHING SYSTEM (PBS)	REQ'D PAYLOAD BAY LENGTH = 8.5 FT, WEIGHT = 565 LB, TRANS COST = $18.38M
OPTION B:	RECONFIGURABLE SATELLITE SERVICING SYSTEM (RSSS)	REQ'D PAYLOAD BAY LENGTH = 5 FT, WEIGHT = 2400 LB, TRANS COST = $11.23M

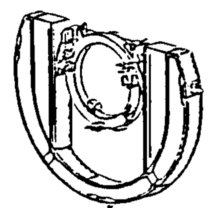
* Transportation costs calculated using standard STS pricing formula and dedicated Shuttle price of $92M (FY'86)

SERVICING HARDWARE DESCRIPTIONS



2ND RMS ARM

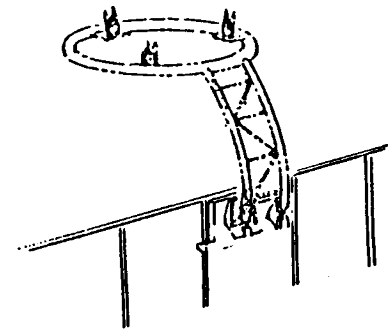
IN-BAY DOCKING FIXTURES



OPTION A: FSS

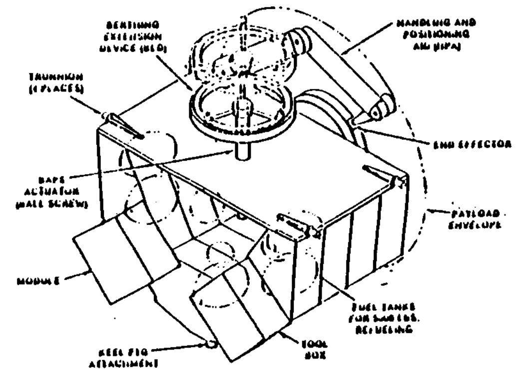
OPTION B: LIGHT-WEIGHT FSS

OUT-OF-BAY DOCKING FIXTURES



OPTION A: PBS

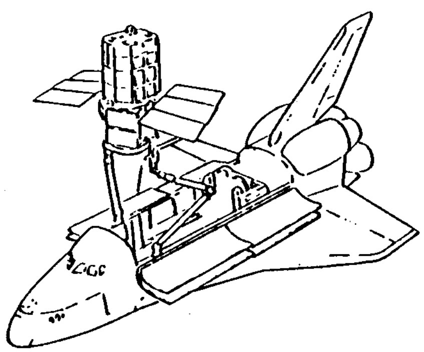
(ROCKWELL INT'L.)



OPTION B: RSSS

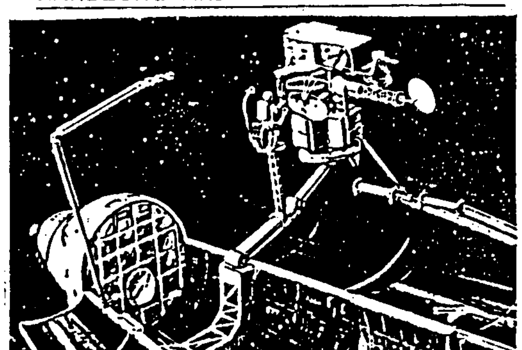
(FAIRCHILD)

HANDLING AND POSITIONING AIDS



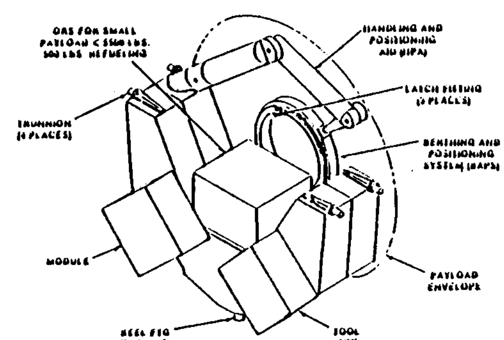
OPTION A: RMS-BASED

(SPAR)



OPTION B: "STRONG ARM"

(GRUMMAN)



OPTIONS C & D: OPTIONS A & B WITH CRADLE

SERVICING HARDWARE OPTIONS

The questionnaire was basically concerned with exchange scenarios and the hardware to be used for the exchange. It was assumed that there would always be at least one RMS arm available in each scenario. To further assist in the payload exchange, the Shuttle Orbiter could have a second RMS arm and/or zero, one or two other docking fixtures as described on the preceding page. A four-digit numbering system was developed to indicate the specific hardware combinations available from the usable options; and the explanation of this numbering system is as follows:

FIRST	- Number of RMS-type arms (1 or 2)
SECOND	- Number of Handling and Positioning Aid-type arms (0, 1 or 2)
THIRD	- Number of in-bay docking fixtures (0, 1 or 2)
FOURTH	- Number of out-of-bay docking fixtures (0, 1 or 2)

Due to the nature of this problem, there will never be a need to carry more than three hardware items on any exchange mission. Thus, if the four digits in the numbering scheme are added together, the total will be three or less. This numbering system will be used on subsequent pages as part of the explanation of questionnaire results.

SERVICING HARDWARE OPTIONS

- ALWAYS AT LEAST ONE RMS-TYPE ARM

- ADDITIONAL HARDWARE OPTIONS INCLUDE:
  - SECOND RMS-TYPE ARM
  - HANDLING AND POSITIONING AID-TYPE ARM
  - IN-BAY DOCKING FIXTURES
  - OUT-OF-BAY DOCKING FIXTURES

- MAXIMUM OF THREE HARDWARE ITEMS TOTAL FOR EXCHANGE MISSIONS

- FOUR-DIGIT NUMBERING SYSTEM DEVELOPED TO INDICATE SPECIFIC HARDWARE COMBINATIONS USED IN EACH SCENARIO

SURVEY DESCRIPTION

In order to gain a sense of what will be generally acceptable methods to accomplish this type of payload exchange, a survey of potential options was constructed. This material was divided into two sections, the first dealing with representative hardware concepts and the second focussing on scenarios to conduct the necessary exchange. Within each section the participant was asked to rate each hardware concept or exchange scenario in three different categories. These categories for the hardware and scenario sections are shown on the facing page. A numerical scale divided from 1 (lowest ranking) to 5 (highest ranking) was used for each category and a weighted average across all three categories was calculated for each item (hardware and scenario). The participants were also asked to match the most useful hardware concept to each scenario in the second section. Results obtained from Fairchild Space Company and RCA AstroElectronics will be discussed on the following pages.

SURVEY DESCRIPTION

- SURVEY MATERIAL DIVIDED INTO TWO SECTIONS
  - (1) REPRESENTATIVE HARDWARE CONCEPTS
  - (2) MODULE EXCHANGE SCENARIOS
- CONCEPTS IN EACH SECTION RATED IN THREE DIFFERENT CATEGORIES
  - (1) HARDWARE
    - IMPORTANCE OF EQUIPMENT TO ANTICIPATED NEEDS
    - FREQUENCY OR DURATION OF EQUIPMENT USE
    - QUALITY OF TOOL AS A SOLUTION TO ANTICIPATED NEEDS
  - (2) EXCHANGE SCENARIOS
    - EFFICIENT UTILIZATION OF STS ASSETS
    - UTILIZATION OF ON-ORBIT PERSONNEL
    - QUALITY OF EXCHANGE SCENARIO AS A SOLUTION TO ANTICIPATED NEEDS
- HARDWARE CONCEPTS MATCHED TO SCENARIOS AS PART OF RATING PROCESS
- PARTICIPANTS: FAIRCHILD SPACE COMPANY AND RCA ASTRO-ELECTRONICS

HARDWARE RESULTS

The facing page lists all hardware examples included in the survey along with the estimated transportation costs and participant response to each. As can be seen, RCA places little value on those systems which use payload bay space since from their perspective this problem can be solved without this equipment. However, RCA does indicate that devices of this type may be useful on a limited basis if the hardware already exists. In contrast it can be seen that a second arm is considered necessary by RCA and of the choices available, a second RMS located on the starboard longeron is preferable.

Fairchild Space Company (FSC) also considers a second arm to be necessary and prefers the structural rigidity offered by the HPA. FSC also foresees the need to carry several ORU's into orbit to repair or enhance the orbiting platform. This spacecraft will also require refueling, the amount and timing of which will depend on the rendezvous altitude used by the Shuttle Orbiter. Both of these items will require supporting structure for launch and landing. As a consequence FSC tends to favor those concepts which include a cradle to which these modules could be attached.

One additional comment should be included here regarding the PBS. This concept was included to represent a means of berthing a payload outside of the orbiter's payload bay envelope. But this particular hardware was applied to a situation for which it was not designed. If this concept were to be applied to a more generic mission set, the specifics of the hardware would undoubtedly be redesigned to reduce transportation costs and increase load capacity.

HARDWARE RESULTS

HARDWARE ITEM	TRANS COST* ($M)	FSC RATING	RCA RATING	COMMENTS
SECOND RMS ON SILL	1.76	2.0	3.5	
HPA ON SILL	3.43	3.3	3.0	
RMS-BASED HPA ON SILL	1.51	2.8	3.0	
HPA WITH CRADLE	9.19	4.3	3.0	CRADLE IMPORTANT FOR CARRYING ORU'S
RMS-BASED HPA WITH CRADLE	9.19	3.3	3.0	CRADLE IMPORTANT; RMS MAY BE WEAK
FSS	6.22	4.3	1.0	
LIGHTWEIGHT FSS	5.11	5.0	1.0	
PBS	18.38	1.5	1.0	CANNOT HOLD PAYLOAD MASS WITHOUT MODIFICATION
RSSS	11.23	4.8	1.0	

* All cost in FY 1986 dollars; based on standard STS pricing formula and dedicated Shuttle price of $92M (FY86)

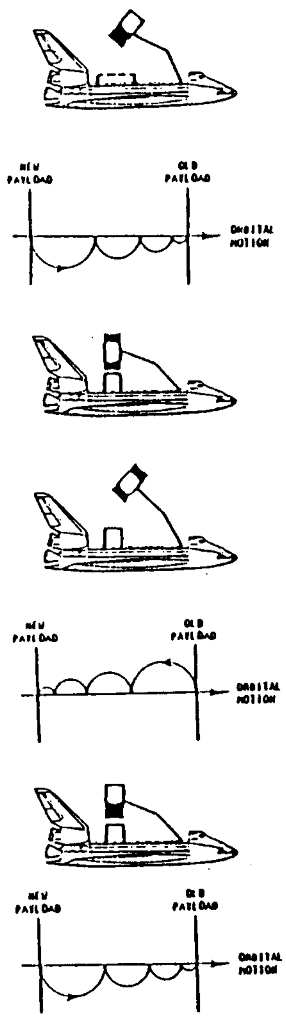
MISSION SCENARIOS FOR ON-ORBIT EXCHANGE OF VERY LARGE PAYLOADS

	MISSION SCENARIO 1	MISSION SCENARIO 2	MISSION SCENARIO 3	MISSION SCENARIO 4, 5*
HARDWARE AVAILABLE	1 RMS ARM	1 RMS ARM AND 1 OTHER DOCKING FIXTURE	1 RMS ARM AND 2 OTHER DOCKING FIXTURES	2 RMS ARMS
SCENARIO** DESCRIPTION	a) DEPLOY NEW PAYLOAD IN FREE DRIFT b) MANEUVER AWAY AND RENDEZVOUS WITH PLATFORM c) BERTH PLATFORM/OLD PAY-LOAD IN PAYLOAD BAY d) REMOVE OLD PAYLOAD AND DEPLOY IN FREE DRIFT e) RETURN TO NEW PAYLOAD f) MATE NEW PAYLOAD TO PLATFORM AND DEPLOY g) RETURN TO OLD PAYLOAD, GRAPPLE AND STOW	a) DEPLOY NEW PAYLOAD IN FREE DRIFT b) MANEUVER AWAY AND RENDEZVOUS WITH PLATFORM c) BERTH PLATFORM/OLD PAY-LOAD IN DOCKING FIXTURE d) REMOVE OLD PAYLOAD AND STOW e) SECURE PLATFORM AND REN-DEZVOUS WITH NEW PAYLOAD f) GRAPPLE NEW PAYLOAD, MATE TO PLATFORM AND DEPLOY	a) GRAPPLE AND SECURE PLATFORM TO FIXTURE 1 b) REMOVE OLD PAYLOAD FROM PLATFORM AND SECURE TO FIXTURE 2 c) SECURE NEW PAYLOAD TO PLATFORM AND DEPLOY d) STOW OLD PAYLOAD IN PAYLOAD BAY	a) GRAPPLE NEW PAYLOAD WITH RMS 1 AND MOVE TO NON-INTERFERRING LOCATION b) BERTH PLATFORM IN PAYLOAD BAY WITH RMS 2 c) DETACH OLD PAYLOAD FROM PLATFORM AND MOVE TO NON-INTERFERRING LOCATION WITH RMS 2 d) ATTACH NEW PAYLOAD TO PLATFORM WITH RMS 1 e) DEPLOY PLATFORM WITH RMS 1 f) STOW OLD PAYLOAD WITH RMS 2

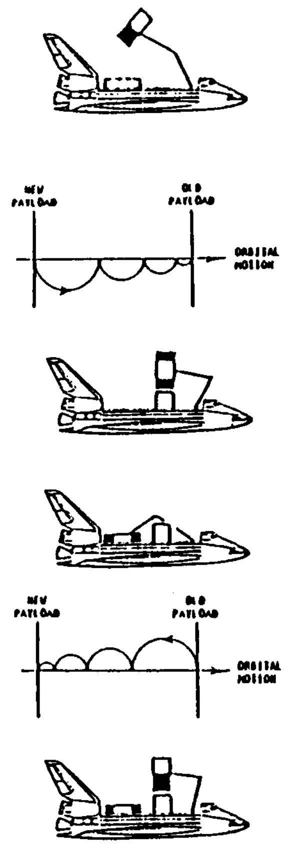
** THE ASSUMPTION WAS MADE IN ALL MISSION SCENARIOS THAT THE PLATFORM CAN BE BERTHED IN THE PAYLOAD BAY, AND THAT THE PLATFORM AND PAYLOADS ARE COMPATIBLE WITH ALL DOCKING FIXTURES

* IN SCENARIO 5 THE PLATFORM IS BERTHED TO A DOCKING FIXTURE RATHER THAN PAYLOAD BAY

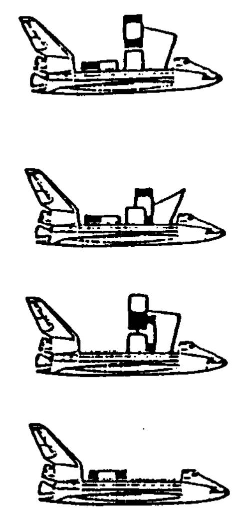
MISSION SCENARIOS FOR ON-ORBIT EXCHANGE OF VERY LARGE PAYLOADS



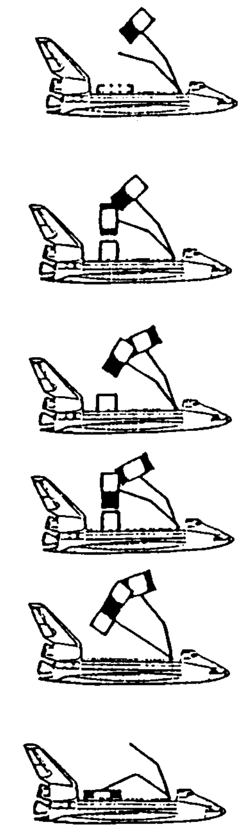
MISSION SCENARIO 1



MISSION SCENARIO 2



MISSION SCENARIO 3



MISSION SCENARIO 4, 5

SCENARIO RESULTS

Of the five scenarios presented, both Fairchild and RCA have discounted the first two primarily due to the fact that one or both of the payload modules would be left unattached to the orbiter and thus significantly increasing the risk to mission success. This is in spite of the fact that these scenarios offer the opportunity to reduce the amount of hardware needed to carry out the exchange. Of the remaining options, those utilizing two arms were the most highly favored as suggested by the ratings in the hardware section. A question was again raised regarding the adequacy of the RMS when left in an "unattended" mode with a large, massive payload still attached.

A difference of opinion exists between the two respondents regarding the worth of a berthing device in the payload bay as part of one of these scenarios. As mentioned previously, Fairchild would use this structure as both a berthing device and as a location to support ORU's while RCA foresees no such needs in its operations. Both agree, however, that if such a device is necessary it should minimize transportation costs.

With these caveats in mind, the best compromise solution appears to be the 1110 case using the port RMS, a sill mounted HPA or RMS-based HPA, and the lightweight FSS.

SCENARIO RESULTS

- ALLOWING EITHER THE OLD OR NEW PAYLOAD MODULE TO DRIFT UNATTACHED SERIOUSLY JEOPARIZES MISSION SUCCESS; ANY SCENARIO RELYING ON THIS OPTION WAS NOT HIGHLY RATED

- THOSE SCENARIOS WHICH UTILIZE TWO ARMS (STANDARD RMS AND EITHER 2ND RMS, HPA OR RMS-BASED HPA) WERE RATED MOST HIGHLY; OF THESE 1110, 1101, AND 2100 WERE RATED HIGHEST

- A QUESTION WAS RAISED REGARDING THE ADEQUACY OF THE RMS WHEN USED AS AN "UNATTENDED" BERTHING DEVICE

- A DIFFERENCE OF OPINION EXISTS ON THE WORTH OF A FIXED BERTHING DEVICE IN THE PAYLOAD BAY, BUT IF SUCH A DEVICE IS USED IT SHOULD MINIMIZE USER COSTS

- ON AVERAGE, THE BEST SCENARIO IS THE 1110 CASE USING THE RMS, A SILL MOUNTED HPA OR RMS-BASED HPA, AND THE LIGHTWEIGHT FSS

SUMMARY

The results from the scenario evaluations indicate that the temporary free drift storage of a payload is not an acceptable option. As a result, additional hardware to be provided by either NASA or the customer will be required to carry out this type of payload exchange. A recommendation which follows from this conclusion is that NASA should establish guidelines which delineate the responsibilities of NASA and the customer regarding the development, ownership and operation of servicing hardware.

At a minimum this additional hardware should include a second arm capability. Potential users would prefer that this arm be installed on the starboard longeron to minimize payload bay usage. A second RMS arm would be the simplest solution to this desire. However, there is some uncertainty among the potential users regarding the adequacy of the RMS as an "unattended" berthing device during that period of time when the single set of controls are used to operate the other arm. The HPA would provide a safer solution to this problem but would be more expensive in terms of development time and cost. As a result, it is recommended that a study of the RMS in its potential role as a berthing device for large, massive payloads be conducted to answer these user concerns.

CONCLUSION	RECOMMENDATION
• FREE DRIFT STORAGE OF PAYLOAD MODULES IS NOT AN ACCEPTABLE OPTION; ADDITIONAL SERVICING HARDWARE IS REQUIRED	• NASA SHOULD PROVIDE GUIDELINES REGARDING SERVICING HARDWARE DEVELOPMENT AND OPERATIONS RESPONSIBILITIES OF NASA AND CUSTOMER
• AT A MINIMUM, A 2ND ARM CAPABILITY IS NEEDED - USERS PREFER TO MOUNT ARM ON STARBOARD LONGERON TO MINIMIZE PAYLOAD BAY USAGE - A SECOND RMS IS SIMPLEST SOLUTION - USER UNCERTAINTY EXISTS REGARDING ADEQUACY OF RMS AS AN "UNATTENDED" BERTHING DEVICE - HPA PROVIDES A SAFER (BUT MORE EXPENSIVE) OPTION	• A STUDY OF THE RMS IN A POTENTIAL ROLE AS A BERTHING DEVICE FOR LARGE, MASSIVE PAYLOADS SHOULD BE CONDUCTED TO ANSWER USER CONCERNS

SUMMARY (CONCLUDED)

In addition to a two arm capability, the survey results indicate that a means of securing the third body to the orbiter is needed. Potential users strongly prefer to minimize the number and/or size of any in-bay berthing device in order to minimize transportation costs. Several possibilities exist to meet this objective. The first is the use of payload bay sill and keel latches as berthing fixtures. In this situation, existing trunnions used for spacecraft launch would be reused as the berthing interface. However, the current latch design should be evaluated for its acceptability in this role. An alternative would be to leave specially designed berthing hardware attached to the spacecraft bus and thus avoid the recurring cost of transporting equipment to and from orbit. Spacecraft designers should examine this option for its effect on overall mission cost.

If an in-bay device is required, the preferred option is the lightweight FSS. This device has potentially the lowest development and transportation cost of the hardware options presented. It also provides a location for the storage of ORU's during launch and landing. Thus, a study of lightweight cradle designs to support the FSS berthing ring is needed.

SUMMARY (CONCLUDED)

CONCLUSION	RECOMMENDATION
• A MEANS OF SECURING THE THIRD BODY TO THE ORBITER (IN ADDITION TO THOSE HELD BY THE TWO ARMS) IS NEEDED	• ACCEPTABILITY OF CURRENT SILL AND KEEL LATCH DESIGN AS PRT OF A BERTHING FIXTURE SHOULD BE ASSESSED
• USERS STRONGLY PREFER TO MINIMIZE THE NUMBER AND/OR SIZE OF ANY IN-BAY BERTHING DEVICES TO MINIMIZE COST - PAYLOAD BAY SILL AND KEEL LATCHES COULD BE USED AS A BERTHING FIXTURE - ALTERNATIVE IS TO LEAVE BERTHING HARDWARE ATTACHED TO SPACECRAFT BUS	• CONCEPTS SHOULD BE DEVELOPED FOR ON ORBIT STORAGE OF BERTHING HARDWARE BY MODIFICATION OF SPACECRAFT DESIGNS TO INCORPORATE THIS FUNCTION
• IF AN IN-BAY DEVICE IS NEEDED, THE LIGHTWEIGHT FSS IS THE PREFERRED OPTION - POTENTIALLY THE LOWEST DEVELOPMENT AND TRANSPORTATION COSTS - ALSO PROVIDES LOCATION FOR ORU STORAGE	• STUDY OF LIGHTWEIGHT CRADLE DESIGN FOR THE FSS BERTHING RING IS NEEDED

REMOTE OPERATING SYSTEMS WITH ASTRONAUT CAPABILITIES

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NOVEMBER, 1985

GRUMMAN

REMOTE OPERATING SYSTEMS WITH ASTRONAUT CAPABILITIES

by Grahme Fischer

Grumman Aerospace Corporation

The group of space based activities that fall within the category of "satellite servicing" are many and diverse. If a small number of machines are to perform all satellite servicing tasks, these machines must be extremely versitile. This paper outlines some characteristics of one such machine, whose abilities are modeled after a human in a space suit performing Extra Vehicular Activity (EVA).

Figure 1 shows a variety of satellites, and lists the locations where their servicing probably will take place. Note the large variety of shapes and sizes of these servicing candidates. Satellite servicing is very different from automated mass production. Instead of repeating the same operation 1000, 10,000 or more times, servicing tasks involve many different tasks (like removing screws, cleaning a lens, connecting electrical or fluid umbillicals, etc.) which are typically repeated fewer than 10 times.

The low repetition rate results from two sources:

- o satellites do not require frequent service, and
- o most satellites are relatively unique, since they are made by different manufactures or design groups without any agreed upon servicability standards.

Examples of the diversity and uniqueness of servicing needs on one large satellite, Space Station, are shown in Figure 2. The table within the figure indicates that the weight of items replaced during servicing (Orbital Replacement Units - ORU) varies by more than two orders of magnitude.

One approach to satellite servicing is to produce a dedicated machine to perform each servicing job. This would require a large number of machines. Another approach, which is probably more efficient, is to build a few different machines (or one universal machine) which can do all servicing jobs. The preceeding paragraph's description of the diversity of servicing jobs argues strongly for the most versitile servicing machine possible.

Whatever types of machines are used for servicing in space, the serviced satellites should also be servicable by an EVA astronaut, if only in a back up mode. This prudent recommendation is made because all systems, no matter how well designed, will malfunction occasionally. Since satellites are expensive high value assets, their abilities to be repaired and restored to effective use have high economic value. Consequently, a second independent mode of repair (EVA) appears highly desireable.

If all servicable satellites are repairable by an EVA astronaut, then a machine which has the same work performing capability as an EVA astronaut will be able to service all those satellites. Such a machine will be extremely versitile, since man, as the most versitile animal, is the standard for high versitility.

The capabilities of an astronaut can be divided into three general categories:

- o manipulation - handling tools, ORUs, satellites and their subsystems
- o sensing - vision, tactile, force and torque
- o intelligence - planning, evaluating, judging

Current state-of-art machinery can be designed which will perform the first two capabilities (for an EVA substitute). However, the production of machinery which duplicates human intelligence is beyond current technology. This difficulty can be overcome by adding a human at the control of the machine system for near term remote operating systems. The most efficient method of controlling dextrous mechanical arms on earth is known as "Telepresence" (The word telepresence denotes a system where the human operator is supplied with as much sensory feedback from the worksite as is practicle to give the operator a sense of being at the worksite.)

The above argument is summarized in Figure 3, "Prudent System Requirements."

Figures 4 and 5 depict two remote operating systems which utilize telepresence. The operators are wearing comfortable clothing in shirtsleeve environments which have been designed to maximize task efficiency. In Fig. 4, the distance between work zones, or the geometry of the work zones, precludes direct vision of the worksite by the astronaut operator. In the system of Fig. 5, Telepresence Work System (TWS), the operator in the Orbiter cabin may or may not have direct visibility to the external worksite. However, the operator is supplied with a full set of sensory data (e.g., video, acoustic, force feedback) from the worksite.

The external working element of an astronaut capable telepresence system is shown schematically in Fig. 6. The Surrogate Astronaut Machine, SAM, has all the dexterity the EVA astronaut has, and, when obtainable at small additional cost, SAM's strength and movement capabilities exceed those of an EVA astronaut. Some of SAM's equipment is shown in greater detail in Fig. 8. SAM has three identical dextrous arms. Two are used as arms and are supported by an anthropomorphic torso. The third arm is used as a device to grab hold of, and lock on to, a worksite. Illumination and television (black and white monocular) cameras are provided on each of the three wrists and at the head. Wrist cameras are fixed in position on the arm, but can provide a variety of views by moving the arm joints. Thus, a monocular head camera provides ample information when "depth perception" type of information can be provided by an orthogonal viewing wrist camera. SAM's dextrous arms and torso (which contains pitch and yaw degrees of freedom) are connected to a transportation device (like an RMS, see Fig. 7) by a strongback. This strongback contains computational and other avionic devices. It also supports a tool box and a battery, which has been sized for a 4 hour mission.

The Telepresence Work System requires a human operator to interpret visual and other sensory feedback from the worksite, and, to direct the movements of SAM. Figure 9 shows a representative control station with an astronaut in an operating position. (The work station in the figure is a ground simulator for a zero "g" station. Consequently, the astronaut is not in the zero "g" relaxed position.)

Astronaut force restraints are provided at the feet, waist and forearms. The waist restraint reduces the magnitude of bending movements transmitted down the astronaut's legs. The foot restraints allow an astronaut to change foot positions and "relock" his feet while operating the system. The resulting change in leg positions should reduce operator fatigue. The forearm restraints prevent cross coupling of inputs from one hand controller to the other.

All components of this control station are stored within the mid-deck cabinets when TWS is not in use.

The 6 degree of freedom (6 DOF) hand controllers of Fig. 9 preclude force feedback to the operator's hands. They are usually associated with rate control of the slave arms (SAM's arms). However, this does not preclude the use of force feedback within the control laws or a force sensory input to the operator (from components other than the hand controllers). The traditional method of supplying force feedback to the operator is through the use of bilateral force reflecting (BFR) replica master arms (arms that the operator "wears" or handles which are kinematically similar to SAM's arms, and produce forces in the master arms which are proportional to the simultaneous forces in SAM's arms). BFR replica master arms are also candidate control devices for TWS.

The TWS concept shown in Figs. 7 through 9 can be operated as a robotic system with supervisory control for many simple tasks. Additional computing hardware and software, over and above that required for telepresence operation, are required to operate roboticly without human intervention. The rest of the system, SAM and the control station, have great utility for robotic operations. Since the SAM can perform all the desired servicing tasks, the external manipulation and sensing systems are adequate for robotic operations. The control station allows an astronaut to perform a "robotic" task for the first few times. The performance of this task, under astronaut control, can "teach" robotic SAM how to perform the task. The computing systems can record SAM behavior as the task is performed and reproduce this behavior for future robotic operations without human intervention. Should a problem arise during robotic operations, the telesensing systems on SAM allow an astronaut, in a supervisory role, to diagnose problems and select appropriate corrective measures. For very difficult problems, the astronaut can take over the task and operate the system in a telepresence mode. This mode brings the maximum amount of human ability to bear on the problem task.

This paper has outlined a logic for designing a machine to perform satellite servicing activities

- o the machine should have at least the same capabilities as an EVA astronaut
- o the machine should operate in the telepresence mode.

Figure 10 summarizes the arguments which have led to the above conclusions. This paper has also outlined the design of a telepresence system that meets the above objectives.

FIG. 1 SOME SERVICING MISSIONS

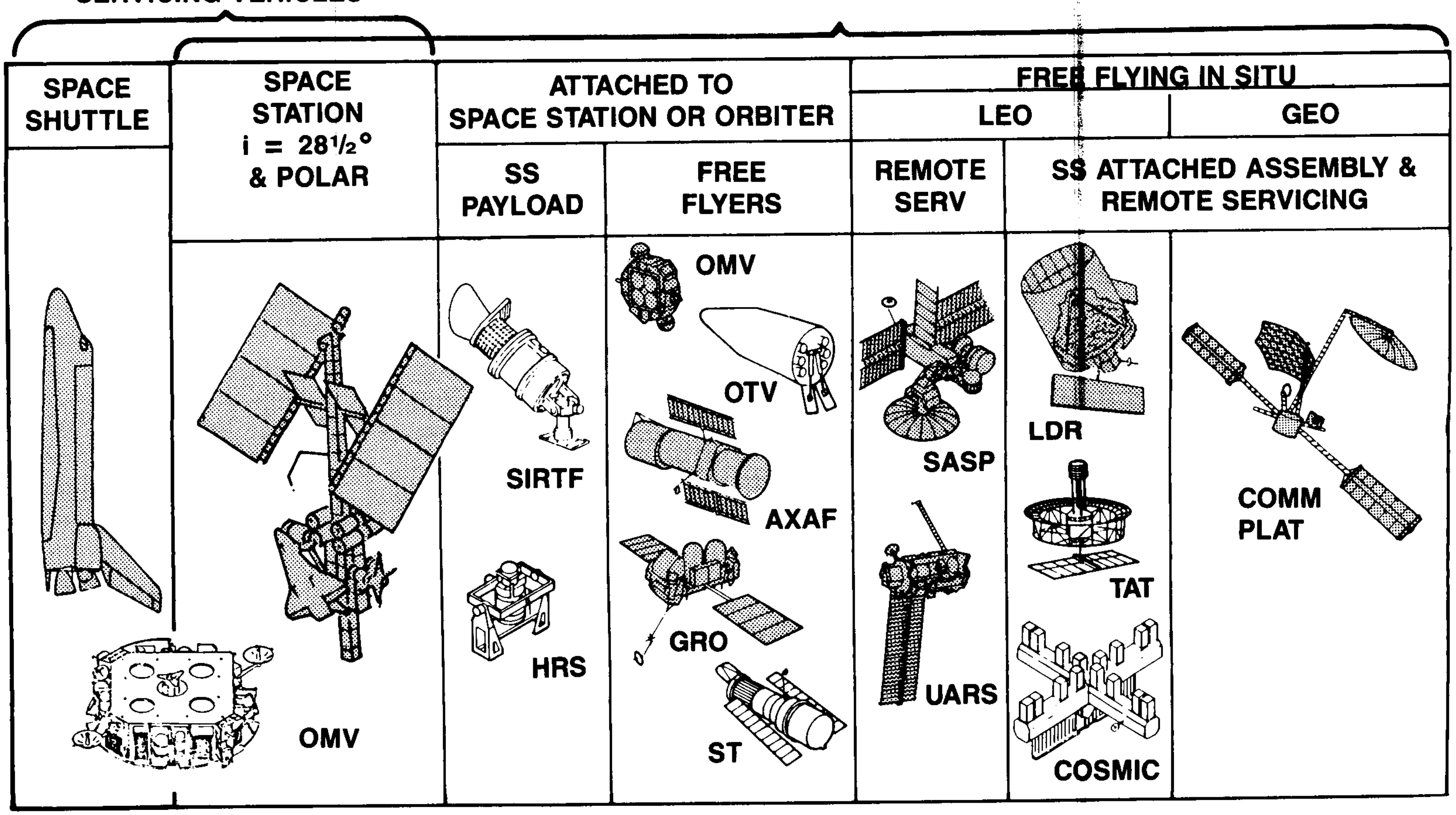

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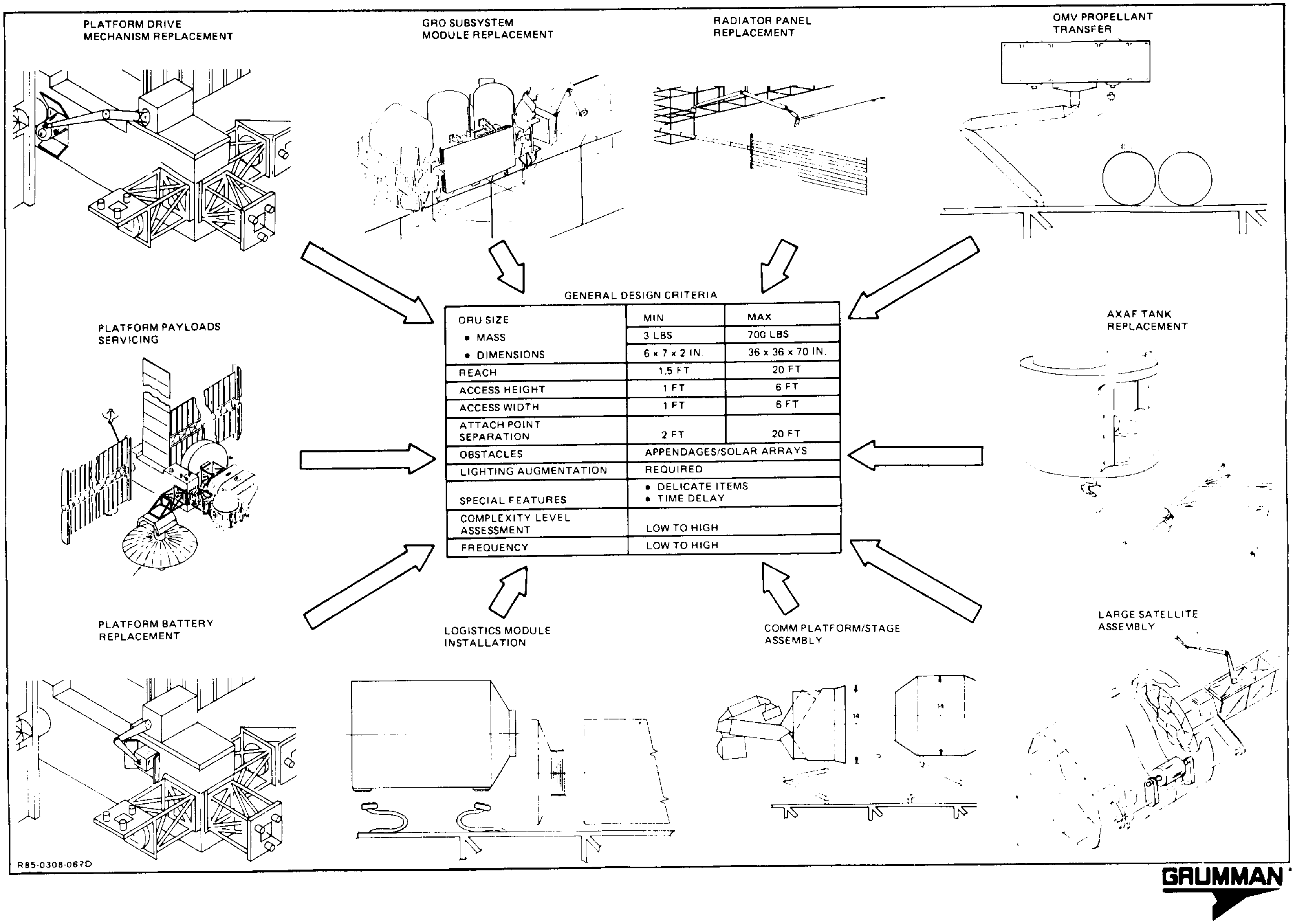
FIG. 2 SPACE STATION SERVICING TASKS



GENERAL DESIGN CRITERIA

	MIN	MAX
ORU SIZE		
• MASS	3 LBS	700 LBS
• DIMENSIONS	6 x 7 x 2 IN.	36 x 36 x 70 IN.
REACH	1.5 FT	20 FT
ACCESS HEIGHT	1 FT	6 FT
ACCESS WIDTH	1 FT	6 FT
ATTACH POINT SEPARATION	2 FT	20 FT
OBSTACLES	APPENDAGES/SOLAR ARRAYS	
LIGHTING AUGMENTATION	REQUIRED	
SPECIAL FEATURES	• DELICATE ITEMS • TIME DELAY	
COMPLEXITY LEVEL ASSESSMENT	LOW TO HIGH	
FREQUENCY	LOW TO HIGH	

FIG. 3 PRUDENT SYSTEM REQUIREMENTS

- ALL SPACE STATION & SATELLITE SERVICEABLE COMPONENTS SHOULD BE DESIGNED TO BE SERVICED BY AN EVA ASTRONAUT
  - AS A MINIMUM, USE EVA IN A BACKUP MODE TO MACHINE SERVICING
  - EXPERIENCE WITH SATELLITE RESCUE & SHUTTLE CREW HAS SHOWN SUBSTANTIAL VALUE FOR OCCASIONAL MANNED INTERVENTION
- MACHINES THAT PERFORM SERVICING SHOULD BE VERSATILE
  - FEW REPETITIVE TASKS
  - LARGE VARIETY OF ACTIVITIES ON A LARGE VARIETY OF NON-STANDARD OBJECTS
- THE MOST VERSATILE MACHINE WILL HAVE, AT LEAST, ASTRONAUT CAPABILITIES
  - MOST VERSATILE ANIMAL IS MAN
  - MACHINE WHICH CAN DO ALL TASKS MAN CAN DO CAN SERVICE ALL SERVICEABLE SATELLITES

- TELEPRESENCE IS REQUIRED
  - CURRENT STATE-OF-ART CAN NOT REPLICATE HUMAN INTELLIGENCE

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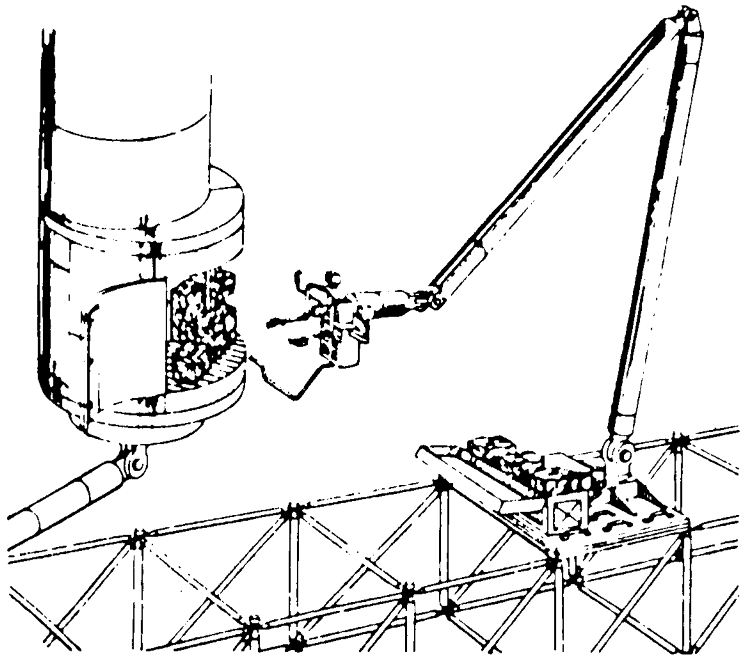
FIG. 4 TELEPRESENCE SYSTEM CONCEPT — SERVICE ON SPACE STATION

SYSTEM ELEMENT	QUANTITY
TRANSPORT VEHICLES	3
POSITIONING ARMS	2 LARGE 1 SMALL
STOWAGE RACKS	2
WORK SYSTEMS	2
CONTROL STATIONS	1 (2 OPERATORS)



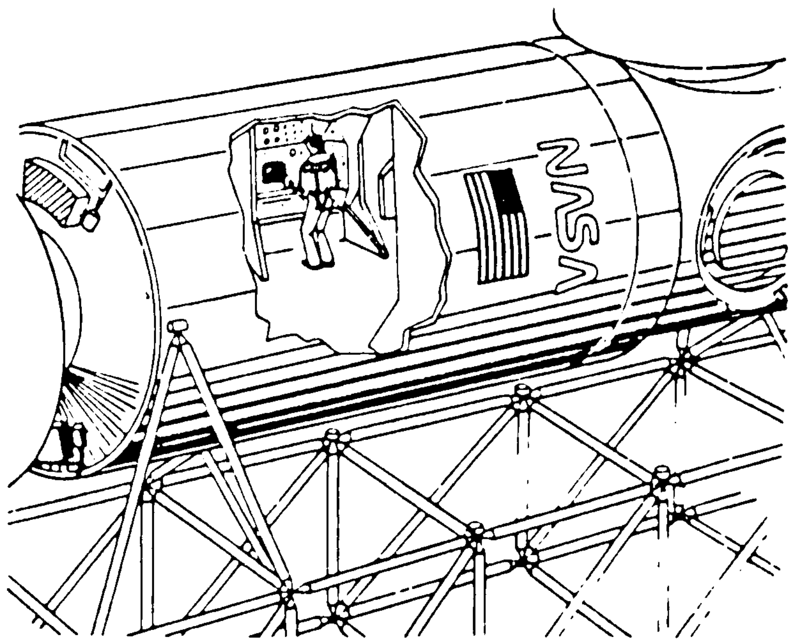


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FIG. 5 TWS — SERVICE ON ORBITER

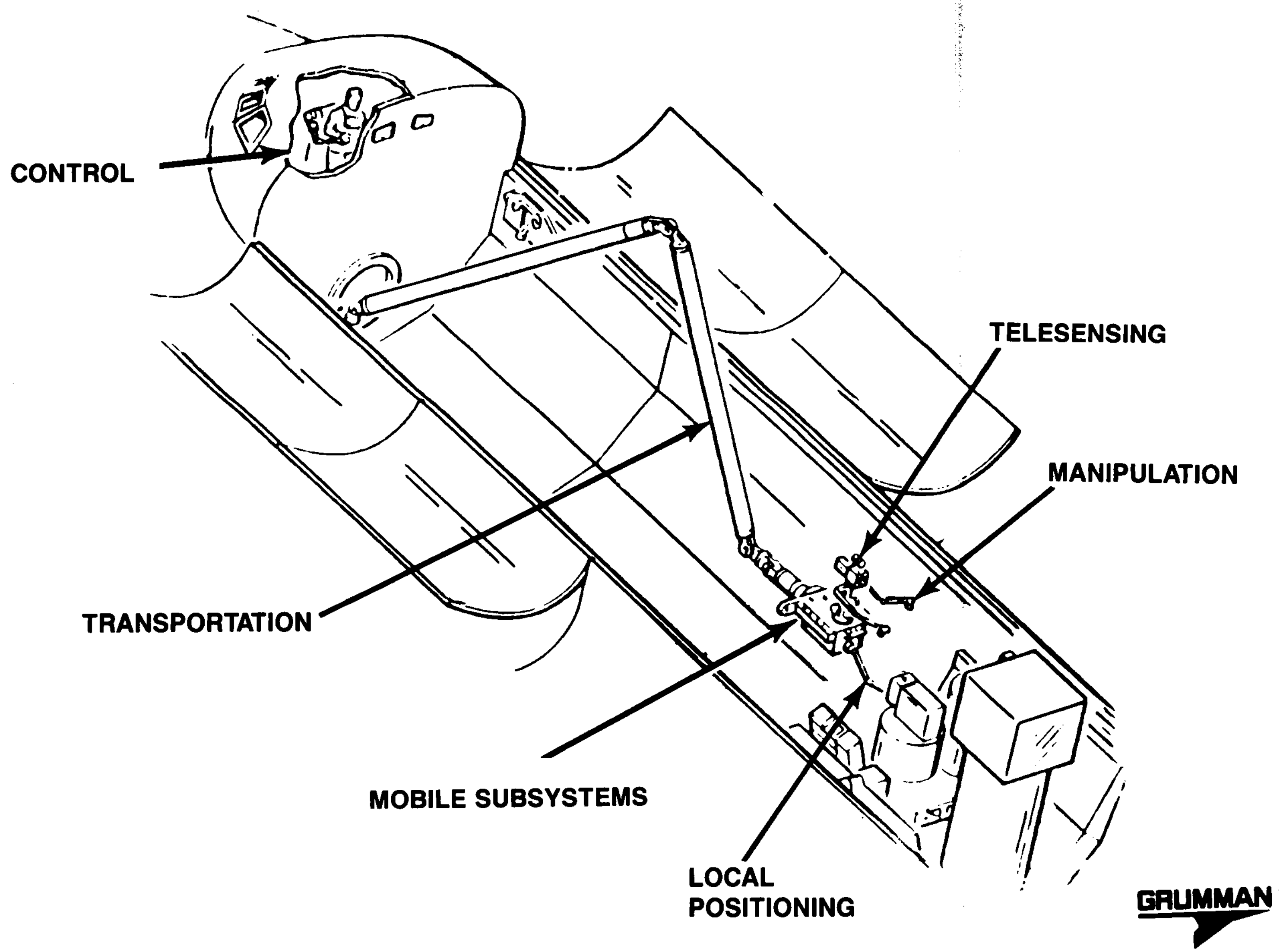

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FIG. 6 PROGRAM GOALS: TWS AND SAM

- DESIGN/UTILIZE MACHINE ELEMENTS TO PRODUCE A TWS THAT IS AT LEAST AS CAPABLE AT PERFORMING WORK AS AN EVA ASTRONAUT

SURROGATE ASTRONAUT MACHINE (SAM)

EVA ASTRONAUT

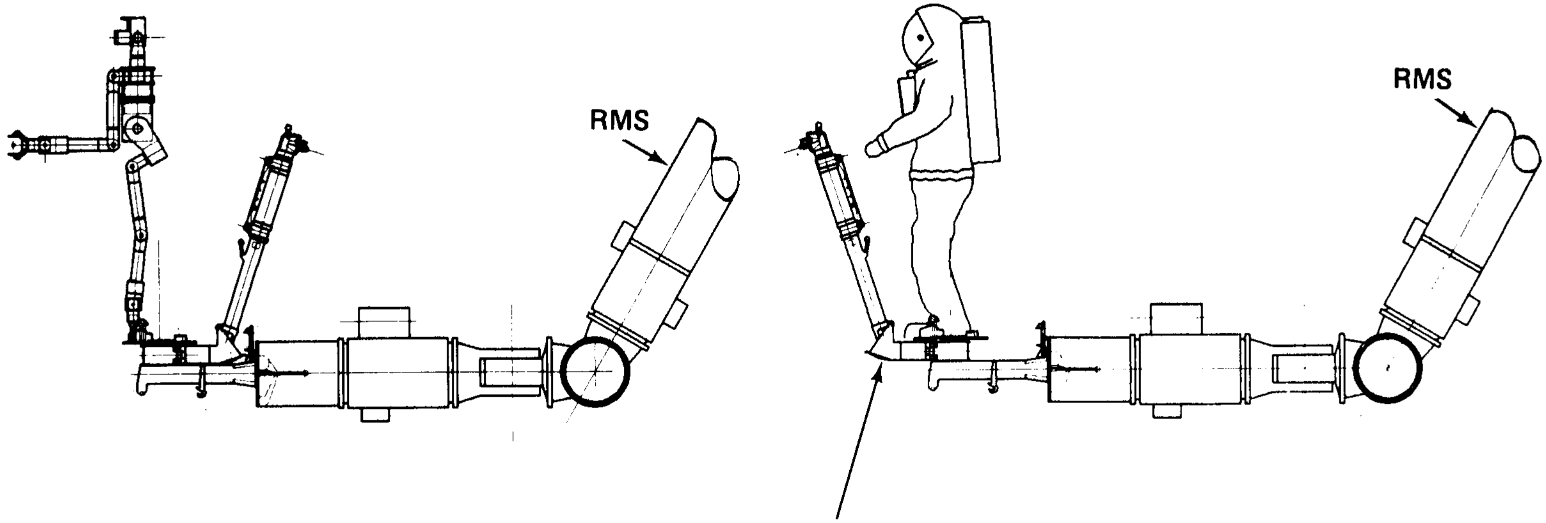

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FIG. 7 TWS EXTERNAL WORKING EQUIPMENT

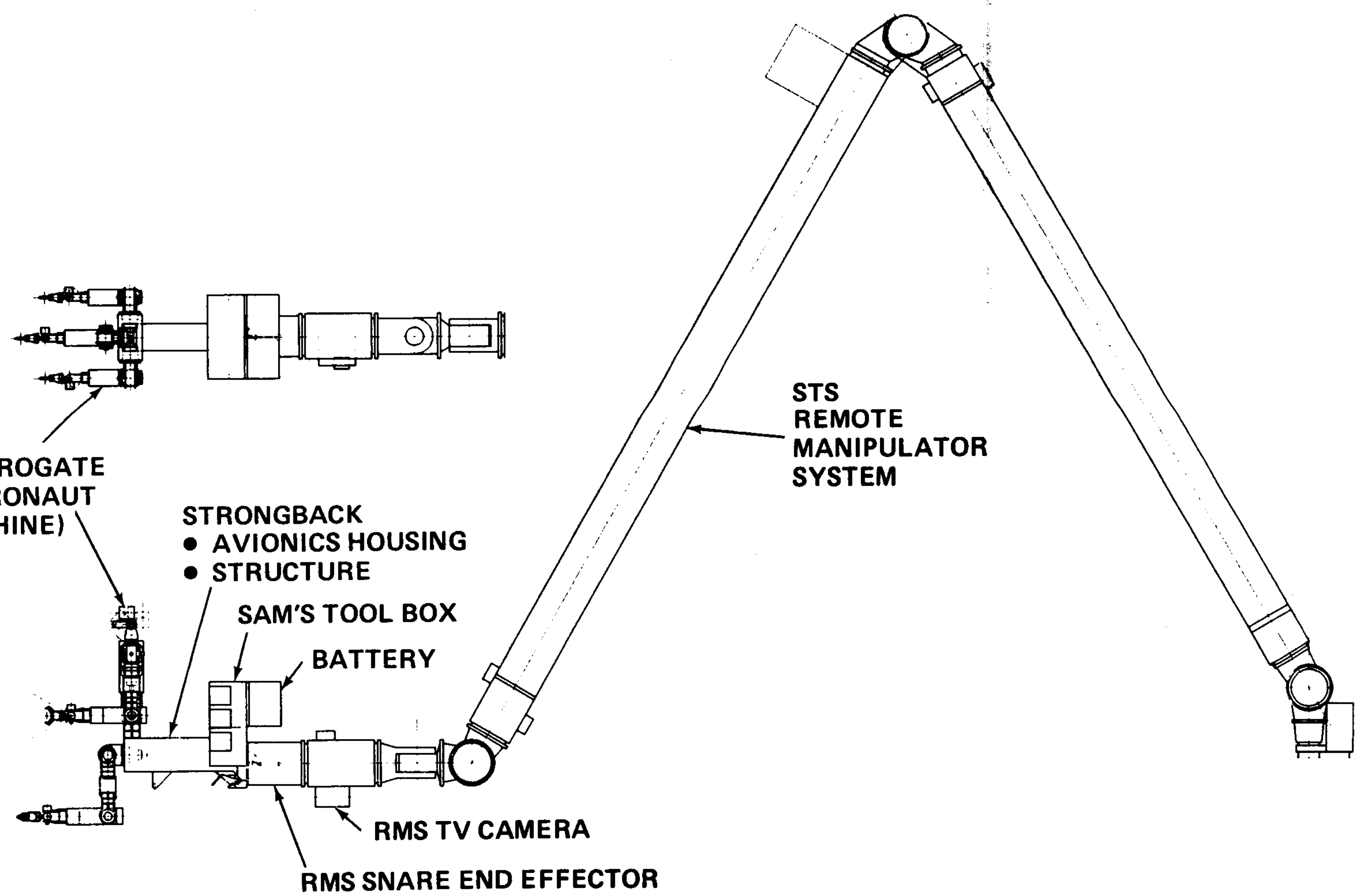

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FIG. 8 SAM AND SUPPORT EQUIPMENT

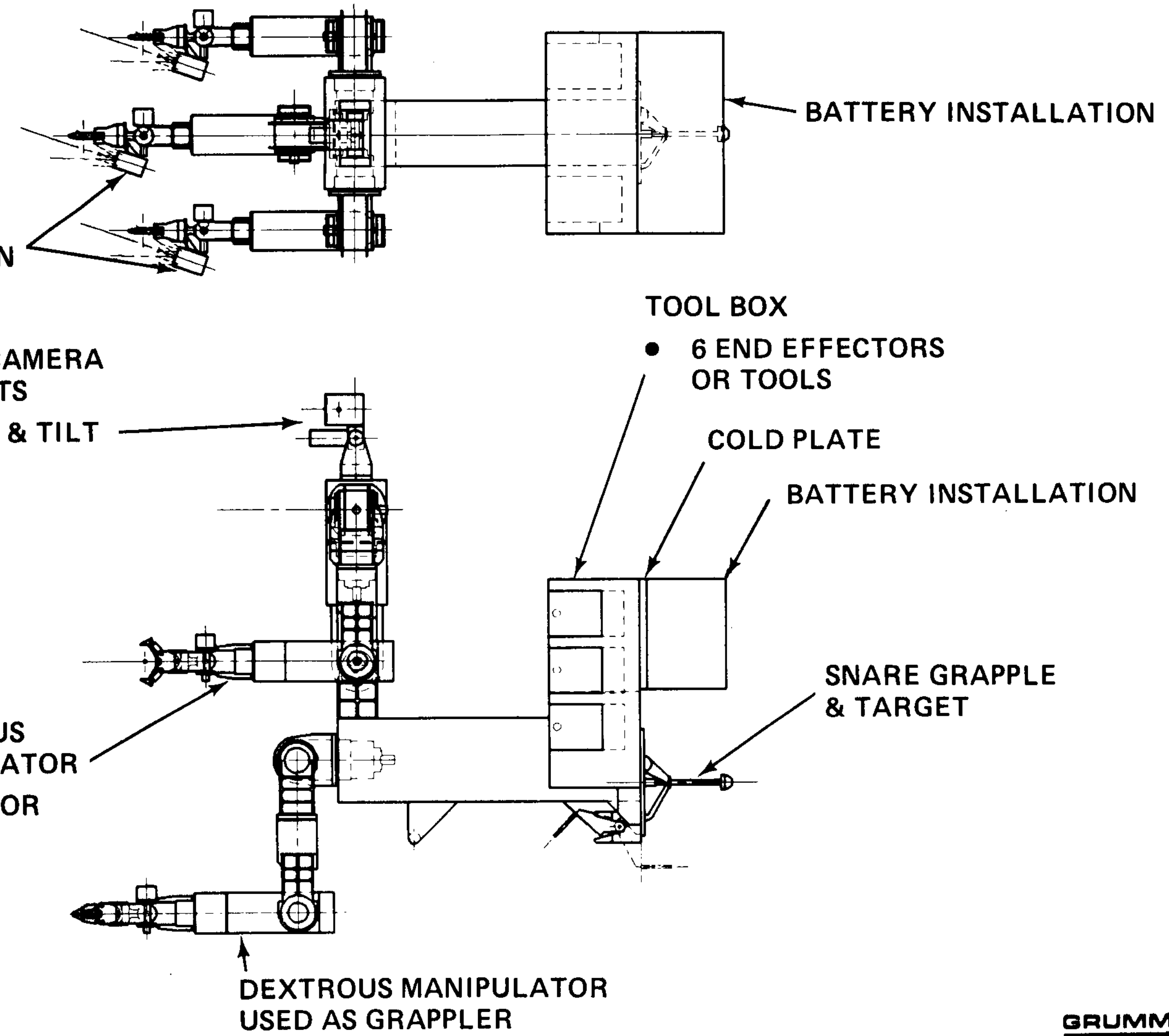

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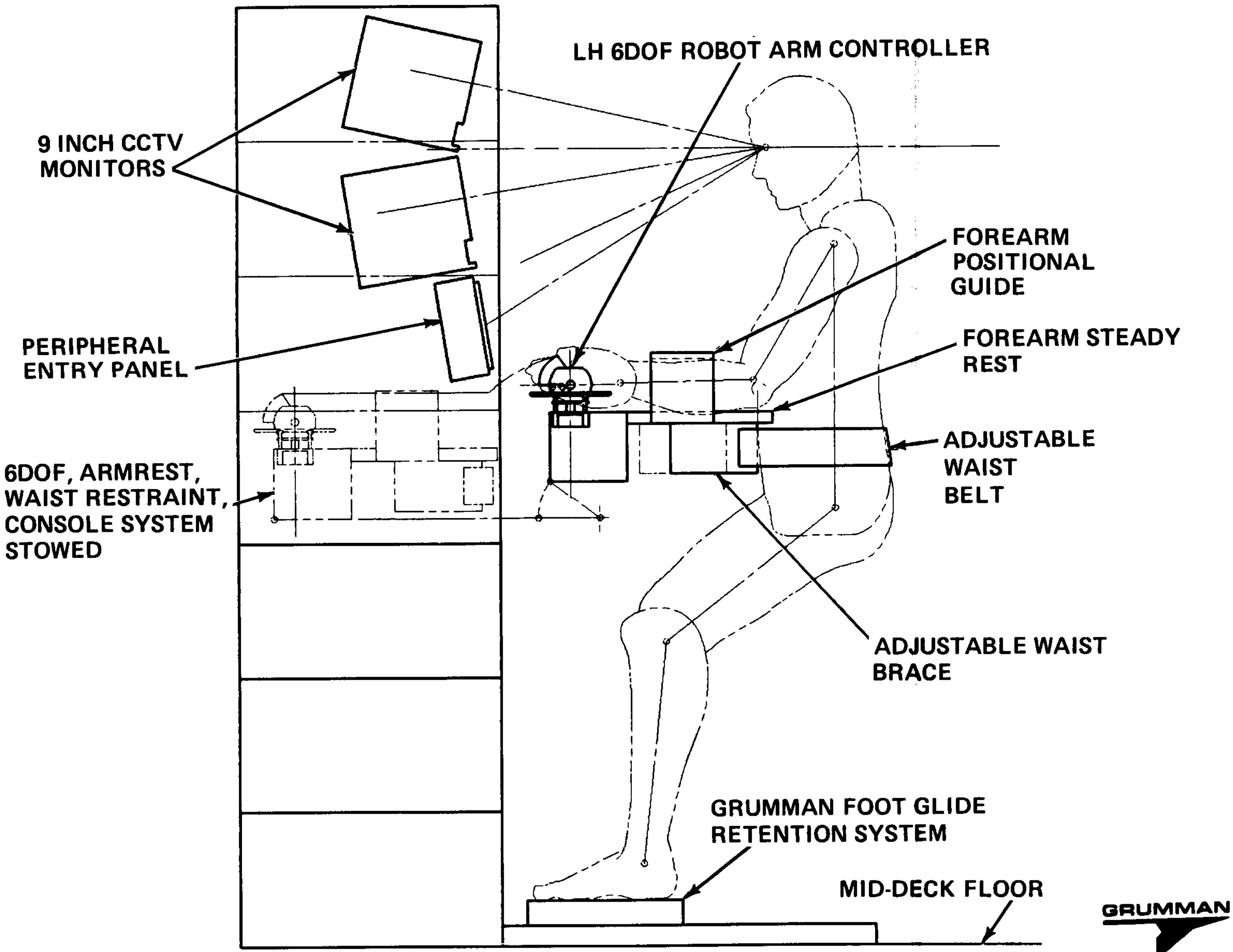
FIG. 9 REPRESENTATIVE TWS CONTROL STATION: SV



SIDE VIEW — SHUTTLE MID-DECK

FIG. 10 CONCLUSIONS

- ALL SERVICEABLE DEVICES SHOULD BE ASTRONAUT SERVICEABLE
- AN ASTRONAUT CAPABLE EVA MACHINE IS WITHIN CURRENT STATE-OF-ART TECHNOLOGY
  - ONLY NORMAL ENGINEERING DESIGN & DEVELOPMENT EFFORT IS REQUIRED
- A SINGLE MACHINE THAT CAN PERFORM ALL SERVICING TASKS IS MOST COST-EFFECTIVE FOR SPACE APPLICATIONS
- HUMAN INTELLIGENCE IS BROUGHT TO A REMOTE WORKSITE WITH TELEPRESENCE
  - AN EVOLUTIONARY STEP TOWARD AUTONOMOUS ROBOTIC SYSTEMS
  - A MEANS TO ALWAYS ALLOW HUMAN INTERVENTION INTO ROBOTIC ACTIVITY
    - o SUPERVISORY CONTROL
  - AN END IN ITSELF FOR VARYING, COMPLEX, HAZARDOUS TASKS

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Dual RMS Application For Satellite Servicing

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Abstract

The Shuttle Orbiter is designed to accommodate two Remote Manipulator Systems which can be installed on the Orbiter port and starboard longerons. So far, only the port RMS has been installed and utilised.

With the starboard RMS installed, not only will the fail-safe capability of the RMS be extended to fail-operational, but also the combined reach envelope and maneuverability of the RMS will be improved considerably. Payloads with deployed appendages (e.g. Solar Maximum Mission) would no longer be required to be berthed in the Cargo bay for servicing.

In this paper, dual-RMS operations are discussed in terms of their benefits and cost, operational sequences, and dynamical interactions between the payload, the RMS arms and the Orbiter.

1.INTRODUCTION.

The Shuttle Remote Manipulator System (SRMS) is designed for dual-arm operations; one arm is installed on the Orbiter port longeron and the other optional arm on the starboard longeron. In order to accommodate the two arm installation, the Orbiter has been designed to provide mounting structures on both longerons, with wiring provisions for power, control and communications between the Orbiter and the starboard arm. The arms can be operated only sequentially. The selection of the active arm is achieved using the SRMS Display and Control Panel; the passive arm is automatically powered off and its brakes are automatically turned on. To date, however, only the port SRMS has been flown on Space Transportation System (STS) missions. The port SRMS has been used to deploy and retrieve payloads successfully from and to the cargo bay, as well as to assist in extravehicular activities and in servicing & repair missions such as Solar Max, Palapa, Westar and Leasat.

With this success, it is natural to think of extending the existing capability of the SRMS to perform a broader class of tasks, thus promoting it from being an element of the Payload Deployment & Retrieval System (PDRS) to a new role in a new application such as assembly and servicing of spacecraft in orbit. To this end, the starboard SRMS would be needed . Applications of dual-SRMS operations are numerous. As an example, during the Solar Maximum Mission (SMM) repair mission, the SMM first had to be berthed and latched in the cargo bay before the SMM servicing could be performed , the SMM deployed solar panels had to be kept clear of the Orbiter radiators and vertical stabiliser during the berthing and re-deployment of the SMM. Had there been the starboard SRMS on-board, the SMM could have been serviced outside the cargo bay in a much simpler manner!

In this paper, we will address the benefits , the cost and the operational issues of the starboard SRMS. First, consider the following scenarios of dual-SRMS operations.

2.DUAL-SRMS OPERATIONS.

Basically, in dual-SRMS operations one arm is used as a support platform to hold a payload while the other arm is used to transport materials between the cargo bay and the payload, or to perform a servicing task on the payload.

2.1. SRMS As A Support Platform.

The Orbiter is first maneuvered to rendezvous with the spacecraft which needs servicing, or the spacecraft is transported to the Orbiter by an Orbital Maneuvering Vehicle (OMV). An SRMS is used to grapple and support the payload whilst the other SRMS is used to conduct servicing operations, for example:

- automatic module exchange without EVA,
- carrying modules to and from spacecraft,
- carrying fluid containers to spacecraft and docking with spacecraft for fluid transfer,
- carrying special purpose end effectors as umbilicals for power and fluid transfer,
- supporting an EVA astronaut on the Manipulator Foot Restraint, etc.

In this fashion one SRMS is used simply as a support and articulation platform for spacecraft servicing. Upon the completion of servicing , the spacecraft is re-deployed by the arm which has been used as the support platform. This operational sequence is depicted by Figure 1.

In a similar scenario, an SRMS can pick up parts of a space structure from the Orbiter cargo bay and hold the part in place for assembly work to be carried out by the other SRMS. Due to its capability to maneuver a payload in six dimensional space, the SRMS can be used to position and orient the payload in a suitable position for assembly. In this example, one SRMS is again used simply as a support platform. Figure 2 illustrates the use of two SRMS's in assembling an Orbital Transfer Vehicle to a payload for geostationary orbit.

2.2. SRMS As A Berthing/Docking Device For Space Station/Platform.

The Orbiter is initially maneuvered to rendezvous with the Space Station/Platform.One SRMS is maneuvered to grapple the grapple fixture on the

Space Station/Platform.Following end effector rigidisation, the SRMS is used to bring the Orbiter close to the Space Station/Platform and support it at a predetermined distance. The other SRMS is then used to transport materials between the Orbiter cargo bay and the Space Station/Platform. In this fashion the SRMS is utilised simply as a berthing/docking device. This scenario may be very beneficial during the Space Station construction phase and Platform resupply. Figure 3 shows a typical operational sequence for the SRMS being used as a berthing/docking device.

2.3. Second SRMS To Support Solar Arrays.

For certain missions requiring longer stay time on orbit and/or higher levels of power consumption than otherwise available, a second SRMS may be used to support a deployed solar array allowing normal operations with the other SRMS. Figure 4 illustrates an SRMS supporting a solar array. The inset shows the power transmission from the solar array to the Orbiter via power cable.

2.4. Second SRMS As A Back-up.

Although the SRMS is specified and designed as a fail-safe system, it would be desirable for certain missions to have fail-operational capability. This capability can be achieved by having two SRMS's on-board. If one arm is out of operation, the other can still be used to complete the mission successfully. Thus, a second SRMS on-board could prove to be beneficial as a back-up for deployment or retrieval of critical payloads.

3.THE BENEFITS OF HAVING TWO SRMS'S ON-BOARD.

Regardless of the scenario described above, it is obvious that the reach capability of the RMS is expanded considerably. Figure 5 shows the reach envelope of the dual-SRMS in comparison with that of a single SRMS. The symmetrical arrangement of the SRMS's with respect to the Orbiter longitudinal plane leads to the fact that the reach envelope of the starboard SRMS is simply a mirror image of the port SRMS reach envelope. Consequently, the reach envelope of a dual-SRMS is symmetrical with respect to the Orbiter longitudinal plane. In this regard, the dual-SRMS is anthropomorphous.

Even within the reach envelope of the SRMS, there are certain points that cannot be reached because of joint angular limits. Figure 6 shows an example of the port SRMS reach capability inside the Orbiter cargo bay at x = 679.5 inches. Again, by symmetry the region that cannot be reached by the port SRMS can be reached by the starboard SRMS. This expanded reach capability undoubtedly would help facilitate the task of storing payloads in the Orbiter cargo bay prior to deployment or after retrieval. Therefore, from an operations viewpoint, the existence of two SRMS's makes handling tasks simpler which is similar to the experience that all of us have in our day-to-day lives.

Furthermore, although the SRMS is designed to be fail-safe and can still operate in Back-up mode in case of primary mode failures, the payload handling task can be carried out more easily using the primary modes of a healthy SRMS. In this connection, a back-up SRMS would upgrade the SRMS capability from fail-safe to fail-operational. During the recent STS 51I mission, a malfunction in the SRMS elbow joint, which occurred early in the flight, forced the arm to be operated in Back-up mode while retrieving the Syncom-3 satellite. The unplanned changes in the astronauts' activities and the longer time line could have been avoided had there been the starboard SRMS on-board!

During the STS 41C mission, the Solar Maximum Mission (SMM) satellite needed to be berthed and latched to the Flight Support System (FSS) in the Orbiter cargo bay prior to its repair. Care had to be exercised during berthing to avoid collision between the spacecraft and the Orbiter. On the other hand, the spacecraft needed to be held in place securely so that repair work could be performed. Once it was latched to the FSS, it could be repositioned/reoriented only in limited manner. Had there been a starboard SRMS on-board, the SMM repair task could have been carried out outside the cargo bay; also the spacecraft could be repositioned and/or reoriented with more freedom. Of course, there are other ways than a second SRMS to provide a stable and maneuverable platform. However, with two identical SRMS's one can use them interchangeably both as a device to deploy and retrieve payloads and as a platform for spacecraft servicing.

With two SRMS's in operation, the astronauts might not need to perform EVA for spacecraft servicing because the payload in service can be maneuvered so that it can be fully monitored by the Orbiter Closed Circuit Television Camera systems and direct viewing through cabin windows. In this connection it is worth mentioning that special tools have been designed to be grappled by the SRMS end effector for remote servicing of payloads. Figure 7 shows the Universal Service Tool which can be used as a remote power screw-driver/socket wrench. Furthermore, a force/moment sensor can be installed on the SRMS end effector to provide feedback to the SRMS operator as to the loads due to contacts between the tool and the spacecraft so that the assembly/service tasks can be performed easily and safely. An SRMS force/moment flight test program is being planned by NASA JSC and JPL with Spar support.

4. THE COST OF OPERATING TWO SRMS's.

There would be no cost to develop the "second" SRMS because currently four such systems are readily available for use, and are interchangeable, port to starboard. If additional SRMS's are required they are basically build to print. This is indeed a strong point for using a second SRMS in comparison with developing, manufacturing and testing a new support system. A second SRMS would not require additional power, other than for thermal control, from the Orbiter because only one SRMS is operational at a time, the other SRMS is powered off.

Carrying a second SRMS on-board obviously incurs a weight penalty cost for launch. However, an SRMS weighs only 450 Kg which is less than the weight of most payloads carried by the Orbiter, and less than one two-hundredths of the

Orbiter weight. Studies conducted for NASA by Science Applications International Corp. have indicated that a second SRMS provides the least transportation cost to orbit compared with other approaches such as the FSS or Reconfigurable Satellite Servicing System. The fact that a second SRMS does not take up any space in the cargo bay and considering the potential benefits it can provide, the weight penalty cost incurred may well be justifiable for many missions.

Although the Orbiter has been built with wiring provisions for power and communications between the Orbiter and the starboard SRMS, the wiring itself has not yet been installed. Therefore, there would be a once only installation cost for this.

Since the starboard SRMS has not been installed and used, it would be necessary to conduct some limited testing on orbit prior to usage. As an example, the control software for the SRMS has been designed to accommodate dual SRMS operations wherein the selection of the active arm is achieved via a port/starboard selector switch on the SRMS Display and Control panel. Similarly, although the astronauts have been trained to operate the port SRMS, they would need additional training to operate the starboard SRMS .Based on human experience of using our own left and right arms, such training would be a relatively minor effort.

5.DUAL-SRMS-OPERATION DYNAMICAL ISSUES.

From a dynamics viewpoint, the Orbiter, SRMS arms and the payload form a closed chain while assembly work and/or spacecraft servicing are carried out. The passive SRMS would absorb the load imposed on it by the active SRMS; for example, loads due to EVA operations or remote module change-out using the Universal Service Tool. Such a load would be transmitted to the Orbiter and fed back through the structure to the active SRMS. The complicated dynamics of the closed chain therefore requires a detailed simulation study of dual-SRMS operations. Currently, there is no real-time or non-real-time SRMS simulation facility which is capable of simulating this type of closed-chain dynamics. In this regard, a study is being carried out at Spar Aerospace Ltd., with funding provided by NASA Johnson Space Center , to upgrade the simulation capability of the real-time (SIMFAC) and non-real-time (ASAD) simulation facilities for evaluation of the closed-chain dynamics problem.

From an operations viewpoint, it is important to determine the holding capability of the passive arm while the active arm is maneuvered with/without a payload ,or while payload servicing is being performed, or while the Orbiter is in a station-keeping mode with its Vernier Reaction Control System (VRCS) in operation. In any of the above situations, loads exist at the six joints of the passive arm. As long as they are below the joint braking torques, the brakes are able to maintain the configuration of the passive arm. If they exceed the joint braking capabilities, the joint brakes will slip to relieve the loads and thus change the arm configuration. Therefore, it is desirable to have sufficiently low loads on the passive arm so that the payload attached to its end effector can be held stationary relative to the Orbiter.

Due to the flexibilty of the arm and the longeron on which the arm is attached, the payload on the passive arm undoubtedly would experience some motions while the active arm is maneuvered or while the Orbiter VRCS is active. The amplitude of such a motion could put constraints on dual-arm operations, such as no VRCS activity during dual-arm operation if the induced flex motion due to VRCS jet firing turns out to be unacceptable or dual-arm operations be carried out only in certain regions within the arm reach envelope so that the stiffness of the passive arm is high enough to attenuate the elastic motion induced on its payload.

An assessment of the impact of the above considerations can only be obtained by computer simulations. In order to have a valid solution the system of Orbiter/Dual SRMS/Payload must be accurately represented in the simulation math model, and also the commands to the SRMS must be realistic. This requires real-time as well as non-real-time simulations. The real-time simulation can also be used to train astronauts for dual-arm operations. These simulations are currently in process at Spar Aerospace Ltd. as part of the above referenced study for NASA JSC.

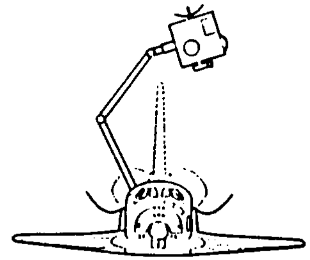
6.CONCLUDING REMARKS.

The benefits of dual-SRMS operations have been discussed, typical operational sequences described, cost benefits outlined and the dynamical interactions between the Orbiter, the SRMS's and the payload addressed. Considering the STS expanded capability made available by the second SRMS such as fail-operational (cf. fail-safe), performing assembly and servicing tasks outside the Orbiter cargo bay (cf. deployment and retrieval of payload only), expanded reach envelope (cf. reach envelope of a single SRMS arm),etc. it is quite clear that the benefits of a dual-SRMS sytem are large in comparison with those for a single-SRMS system. Also, with no additional cost for developing, manufacturing and verifying, a second SRMS is indeed an economically viable solution to the problem of providing a stable platform for assembly/repair of space structures/ spacecraft. These studies are currently being conducted at Spar Aerospace Ltd. to address the operational issues involved.

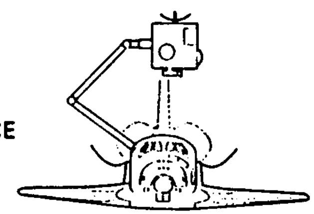
ACKNOWLEDGEMENTS

The author wishes to thank NASA Johnson Space Center and Spar Aerospace Ltd. for the permission to publish this paper. He also would like to thank Messrs. S.S.Sachdev and B.R.Fuller of Spar Aerospace Ltd. for giving helpful comments.

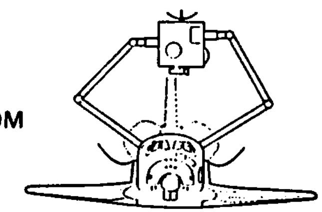
1. RMS 1 GRAPPLES SPACECRAFT



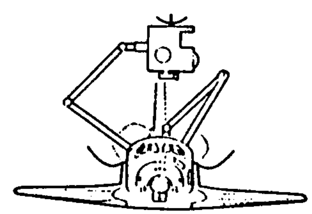
2. RMS 1 BRINGS SPACECRAFT TO SERVICE LOCATION, THEN IS POWERED OFF



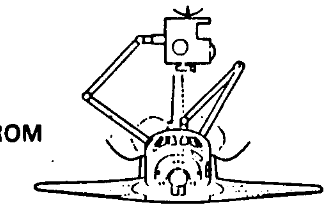
3. RMS 2 REMOVES FAULTY MODULE FROM SPACECRAFT



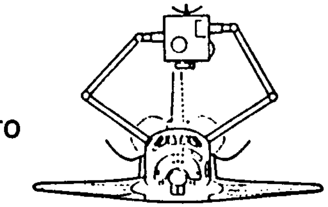
4. RMS 2 BERTHS FAULTY MODULE IN CARGO BAY



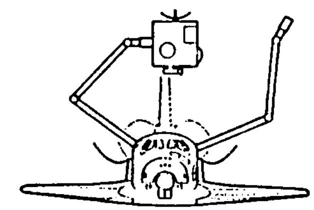
5. RMS 2 PICKS UP NEW MODULE FROM CARGO BAY



6. RMS 2 INSERTS NEW MODULE INTO SPACECRAFT



7. RMS 2 BACKS OUT, THEN IS POWERED OFF



8. RMS 1 IS POWERED ON, THEN REDEPLOYS SPACECRAFT

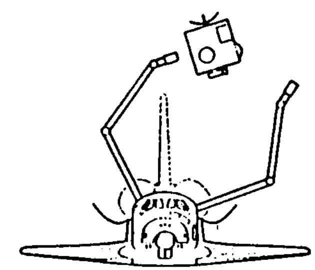


FIGURE 1 MODULE EXCHANGE DUAL-RMS OPERATION

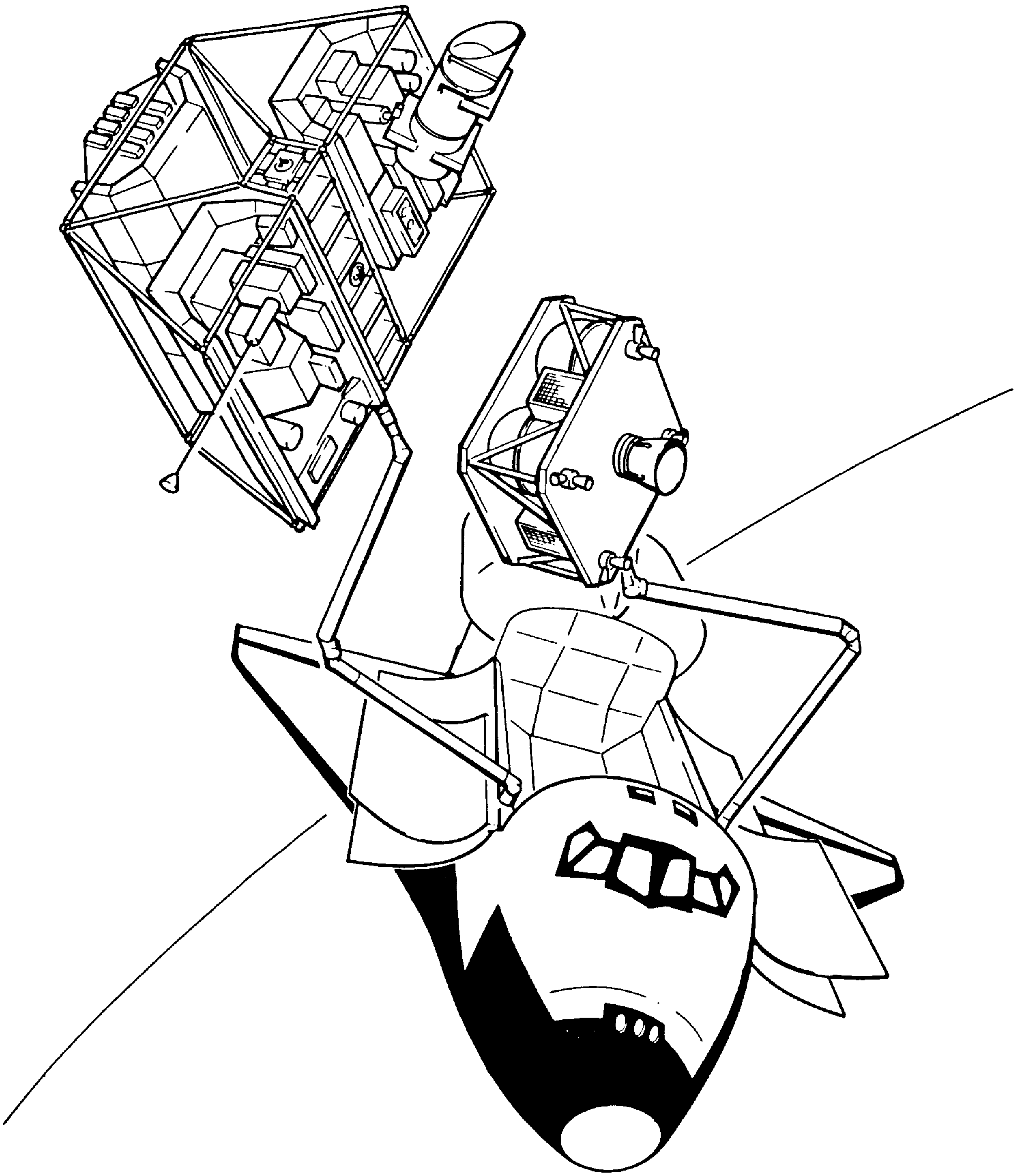
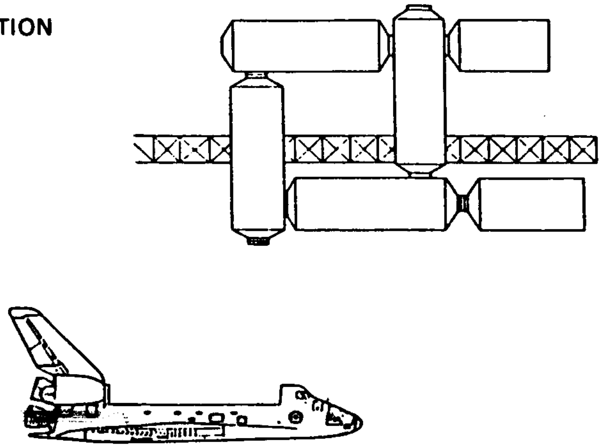
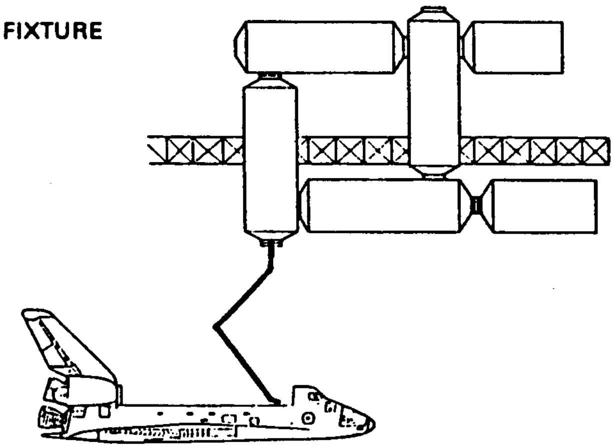


FIGURE 2 ASSEMBLY OF ORBITAL TRANSFER VEHICLE TO PAYLOAD FOR GEOSTATIONARY ORBIT

1. RENDEZVOUS WITH SPACE STATION



2. RMS GRAPPLES THE GRAPPLE-FIXTURE ON SPACE STATION



3. RMS IS MANEUVERED TO BRING ORBITER CLOSE TO SPACE STATION, THEN IS POWERED OFF

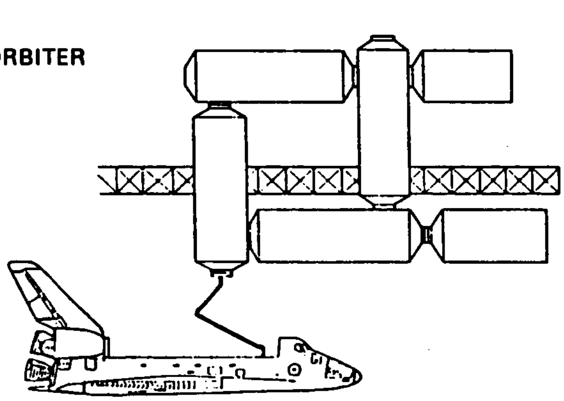


FIGURE 3 RMS AS A BERTHING/DOCKING DEVICE

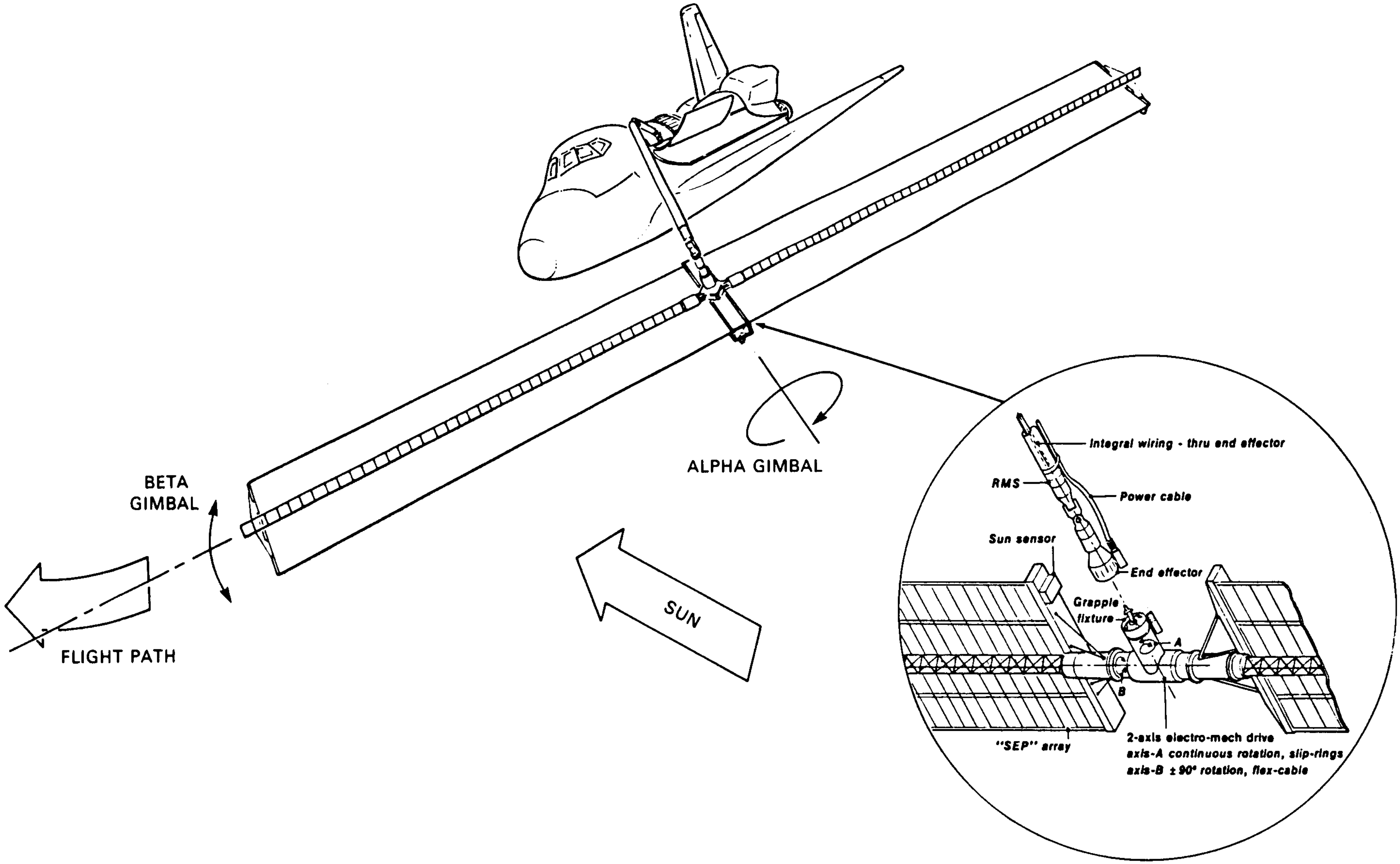


FIGURE 4 SRMS SUPPORTING A SOLAR ARRAY

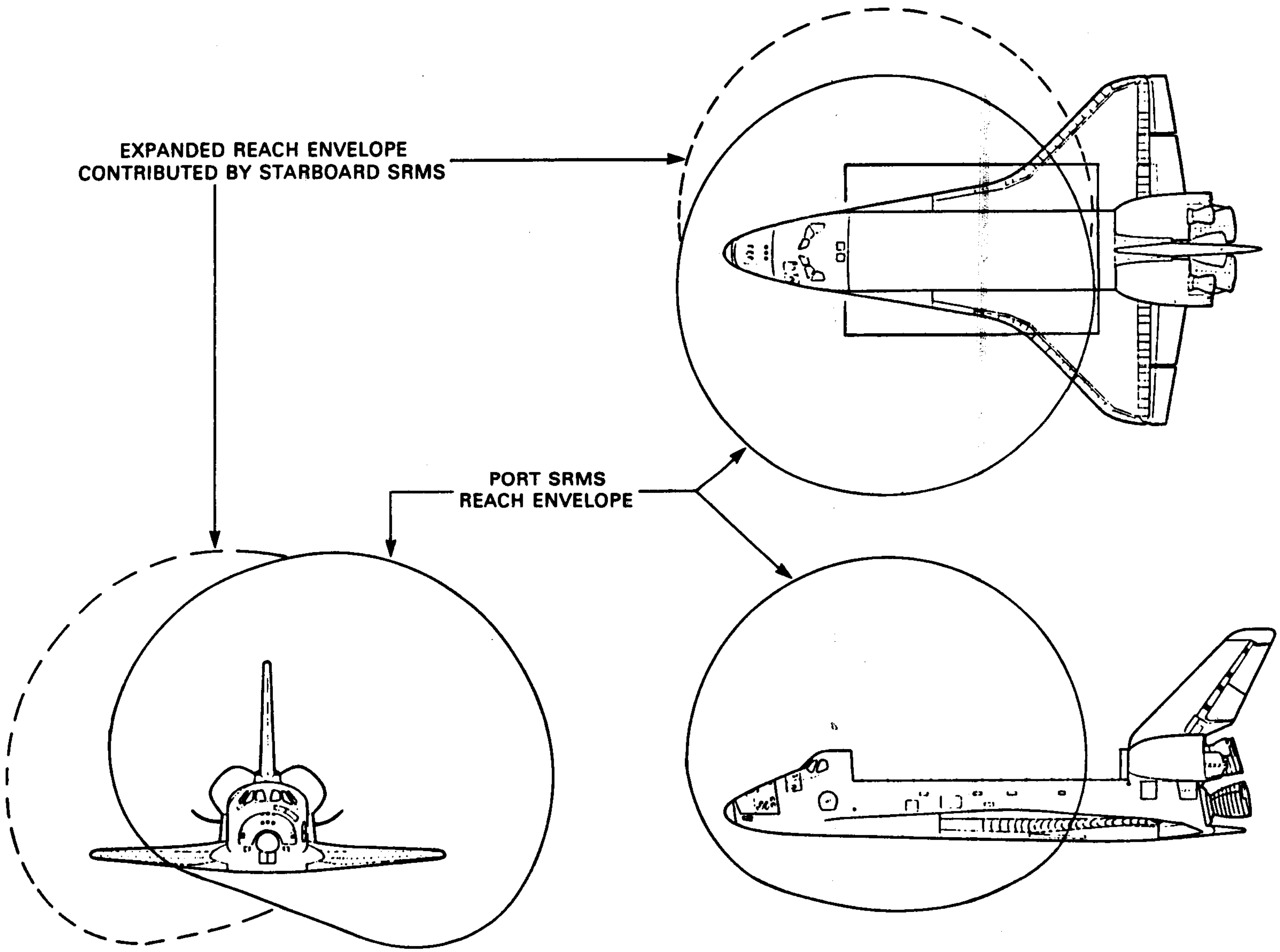


FIGURE 5 REACH ENVELOPE OF DUAL SHUTTLE REMOTE MANIPULATOR SYSTEM

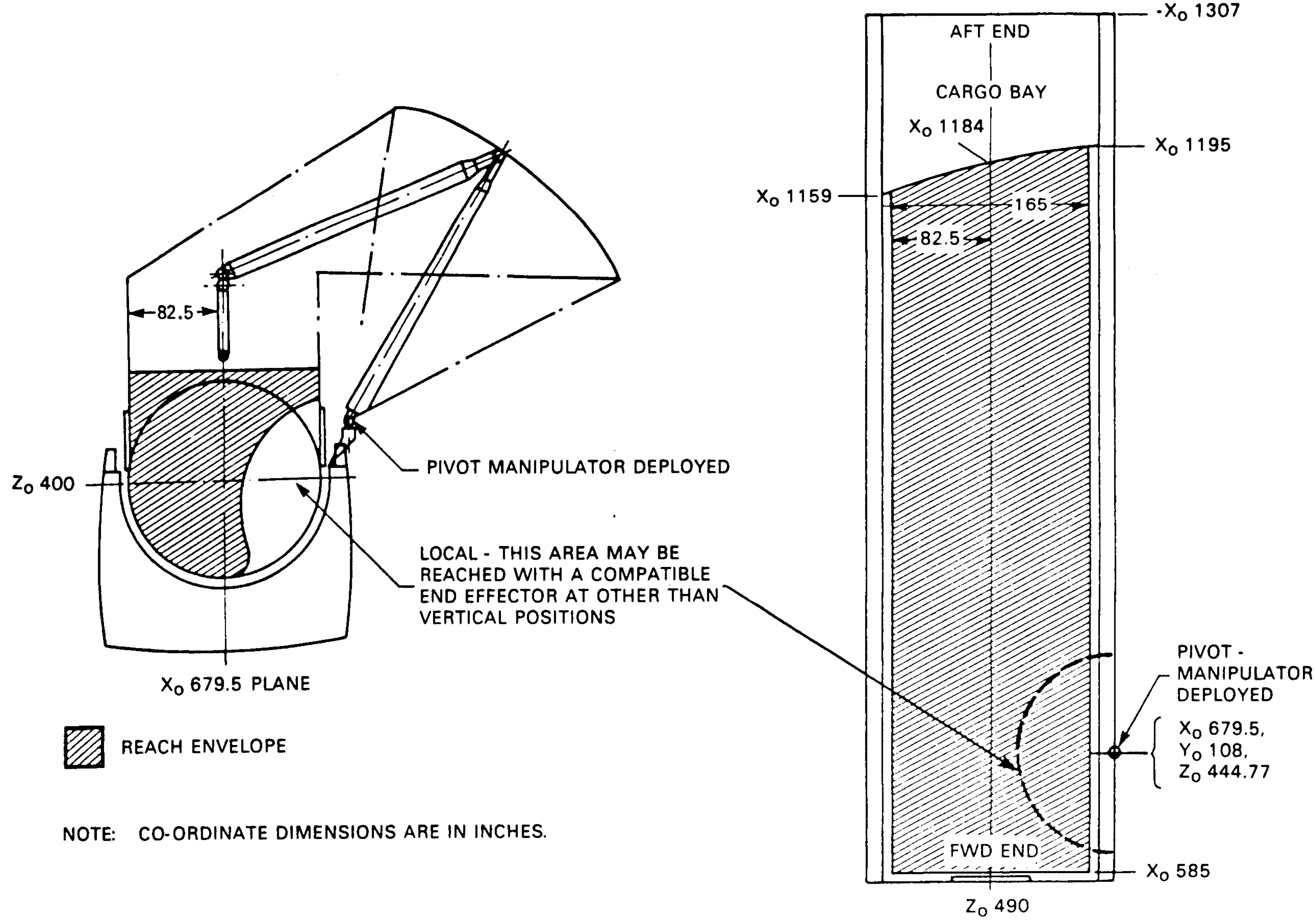


FIGURE 6 REACH CAPABILITY OF PORT SRMS IN ORBITER CARGO BAY

RMS SERVICING CONFIGURATION

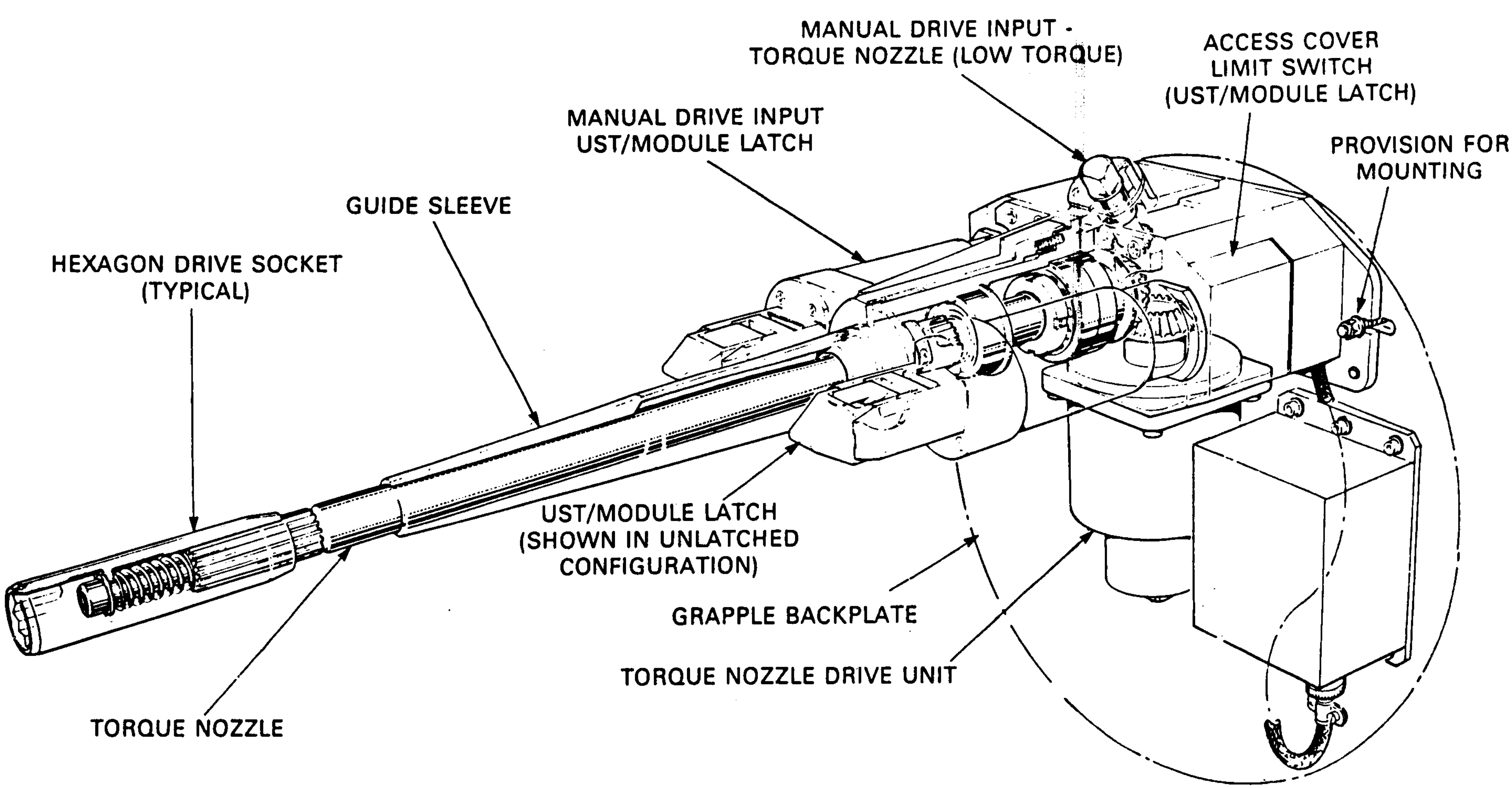


FIGURE 7 UNIVERSAL SERVICE TOOL TO BE GRAPPLED BY SRMS END EFFECTOR

REMOTE SATELLITE SERVICING

DON R. SCOTT

MARSHALL SPACE FLIGHT CENTER

REMOTE SATELLITE SERVICING

DONALD R. SCOTT
MSFC, AL

INTRODUCTION

Studies by TRW and Martin-Marietta have shown, and previous flight experience has indicated, that satellite servicing in orbit is not only feasible, but economical. With the advent of the orbital maneuvering vehicle, a permanent space station and other vehicles that remain in orbit for many years, on-orbit servicing will be a necessity. The viability of local EVA maintenance has been clearly demonstrated by the successful Solar Maximum Repair Mission and the Syncon 3 jumpstart. The concept envisioned here is to extend man's capability through remote servicing where it is impractical or unsafe for EVA. One concept to achieve remote servicing would be to fly a robotic servicer system on a free flyer such as the OMV in which the OMV would dock with a spacecraft requiring maintenance and perform servicing by remote operation.

The MSFC has been performing work in the teleoperator and robotics area for a number of years in anticipation of future needs. This has resulted in unique simulation capabilities, payload servicing concepts including breadboard hardware, robotic arms, end effectors, and rendezvous and docking test bed. The purpose of this effort is to investigate and determine through simulation testing and analysis the performance, constraints, and limitations of the servicing system. Pertinent to this is the evaluation of manipulator arms, remote control stations, visual systems, control modes, end effectors, interface mechanisms, and other aspects that comprise the total servicing system.

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REMOTE SATELLITE SERVICING

The general items listed on the opposing page are believed to be some of the more important tasks that will be required. For a detailed listing of manipulator movements and tasks, the reader is referred to the MIT document "Space Applications of Automated, Robotics and Machine Intelligence Systems (ARAMIS) - Phase II," contract NAS8-34381, dated October 1983, and "Teleoperator Human Factors Study" by MMC contract NAS8-35184, dated January 1984.

Skills required to perform these tasks are rendezvous/docking, viewing, arm positioning, module exchanges, tool handling latch/unlatch manipulation, stabilizing and fluid/electrical umbilical couplings. However, to maximize the utility of the skills above, certain provisions and guidelines should be incorporated in the design of the payloads to be serviced.

REMOTE SATELLITE SERVICING

REQUIRED CAPABILITIES FOR REMOTE SATELLITE SERVICING

- FLUID TRANSFER
- MODULAR EXCHANGES
- INSPECTION
- DEPLOY/RETRACT APPENDAGES
- MAINTAIN/REPAIR
- CONSTRUCTION/ASSEMBLY
- STABILIZE ATTITUDE
- CONTINGENCY CASES

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REMOTE SATELLITE SERVICING

Realistic simulations are paramount to the successful design and development of remote servicing capability. MSFC has been involved for the past 14 years in a research and technology effort that has evolved into the unique simulation capability existing today. Two independent manipulator simulations basically provide automated and man-in-the-loop operation. An air-bearing mobility unit floating on a flat floor and simulating a 6 DOF free-flying vehicle can be associated with and support the manipulator simulations. Technology investigations follow a progression from subsystem level to system level investigations to technology readiness demonstrations that include human factors.

MSFC has been designing and developing satellite servicing subsystem technologies for a number of years, making improvements, and tying all the subsystems together to make a system simulation.

REMOTE SATELLITE SERVICING

MSFC SIMULATIONS ARE SYSTEMS ORIENTED AND INCLUDE ALL SUBSYSTEMS THAT MAKE UP THE ENTIRE SERVICING SYSTEM

- MANIPULATOR ARMS
- CONTROL STATION
- INTERFACE MECHANISMS
- SENSORS
- END EFFECTORS
- SOFTWARE
- VISION SYSTEMS
- TASK BOARDS

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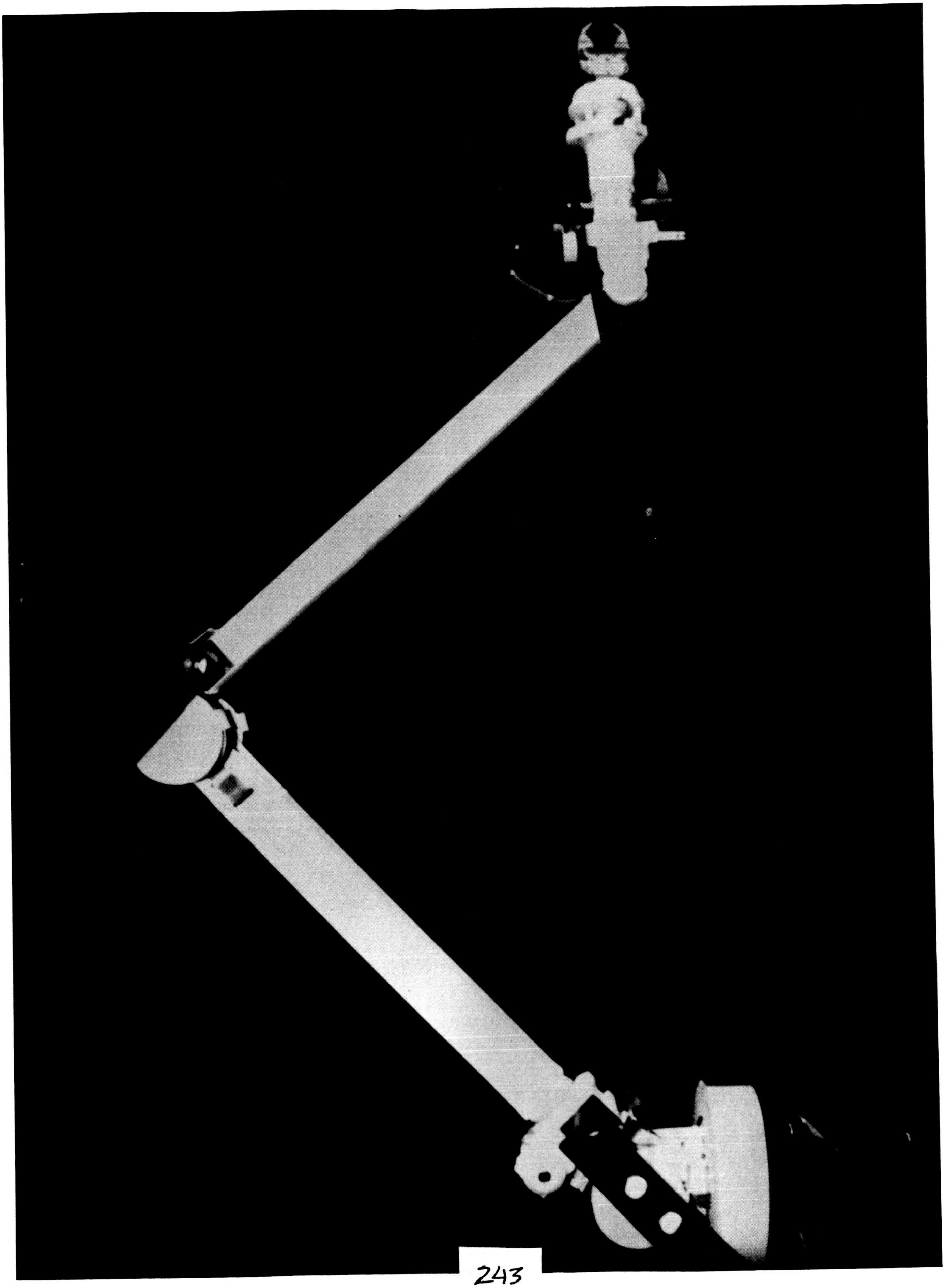
REMOTE SATELLITE SERVICING

PROTOFLIGHT MANIPULATOR ARM

The protoflight manipulator arm (PFMA), a seven-degree-of-freedom arm, was designed for such tasks as satellite servicing and space structure assembly. The PFMA has been exercised in a configuration of cameras, lighting, training task boards, interface console and digital computer.

The PFMA is an anthropomorphic manipulator assembly having flexible joints for shoulder, elbow, and wrist. The shoulder is capable of movements in the pitch and yaw axes. The elbow is capable of pitch movement, with roll/indexing capability between the shoulder and elbow. The wrist assembly provides roll, pitch, and yaw positioning for the end effector. The reach of the entire manipulator is in the range of 25 cm (10 in.) minimum to 200 cm (96 in.) maximum measured along a line from the shoulder pitch axis to the wrist pitch axis. Total arm length including wrist and end effector is 3.05 m (10 ft.). The indexing motion extends coverage to an approximate hemispherical shape over the grasping interface. Each joint consists of a 28 Vdc reversible motor, and movement is accomplished through a system of gears and/or clutches.

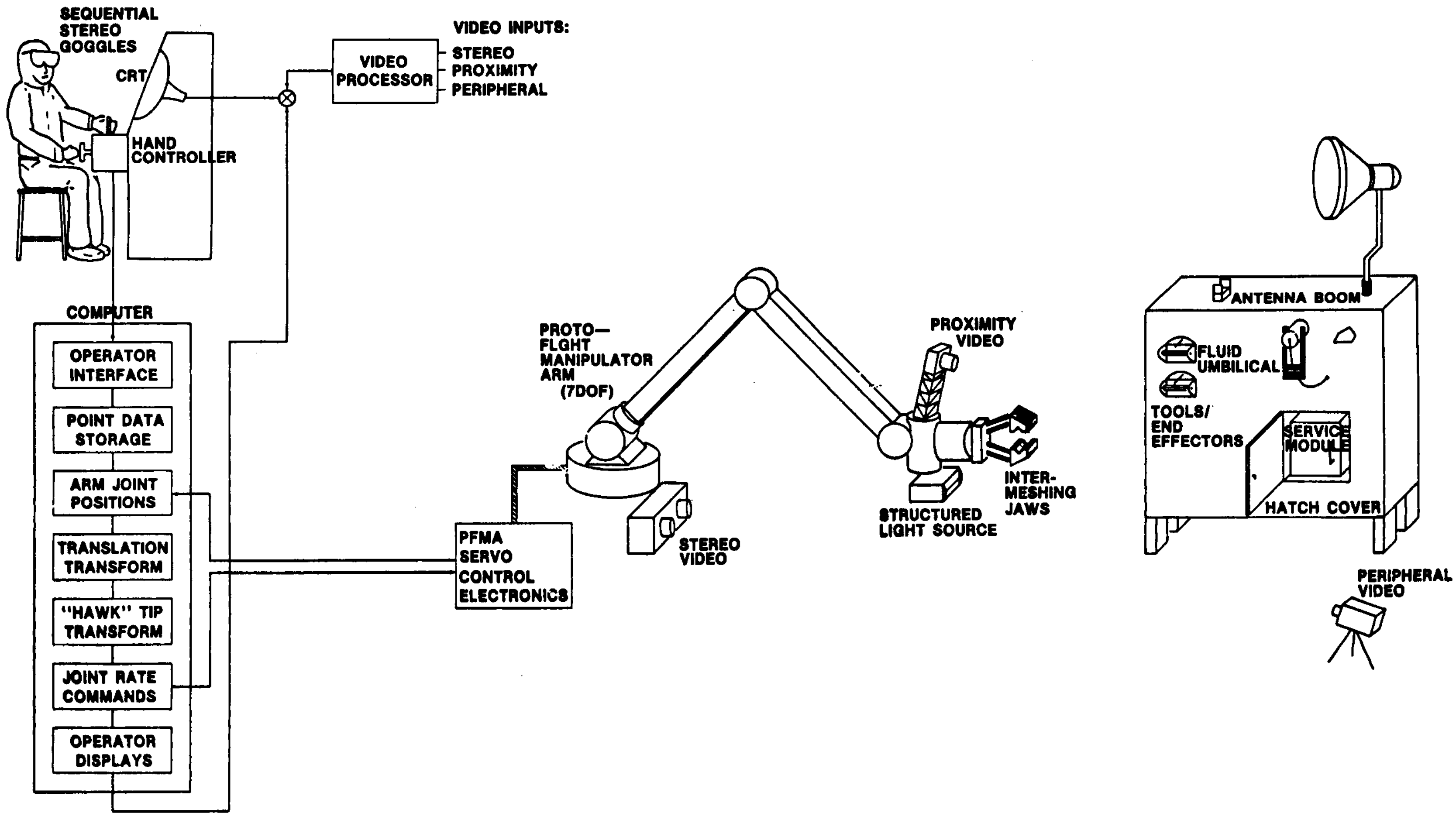


4-0

REMOTE SATELLITE SERVICING

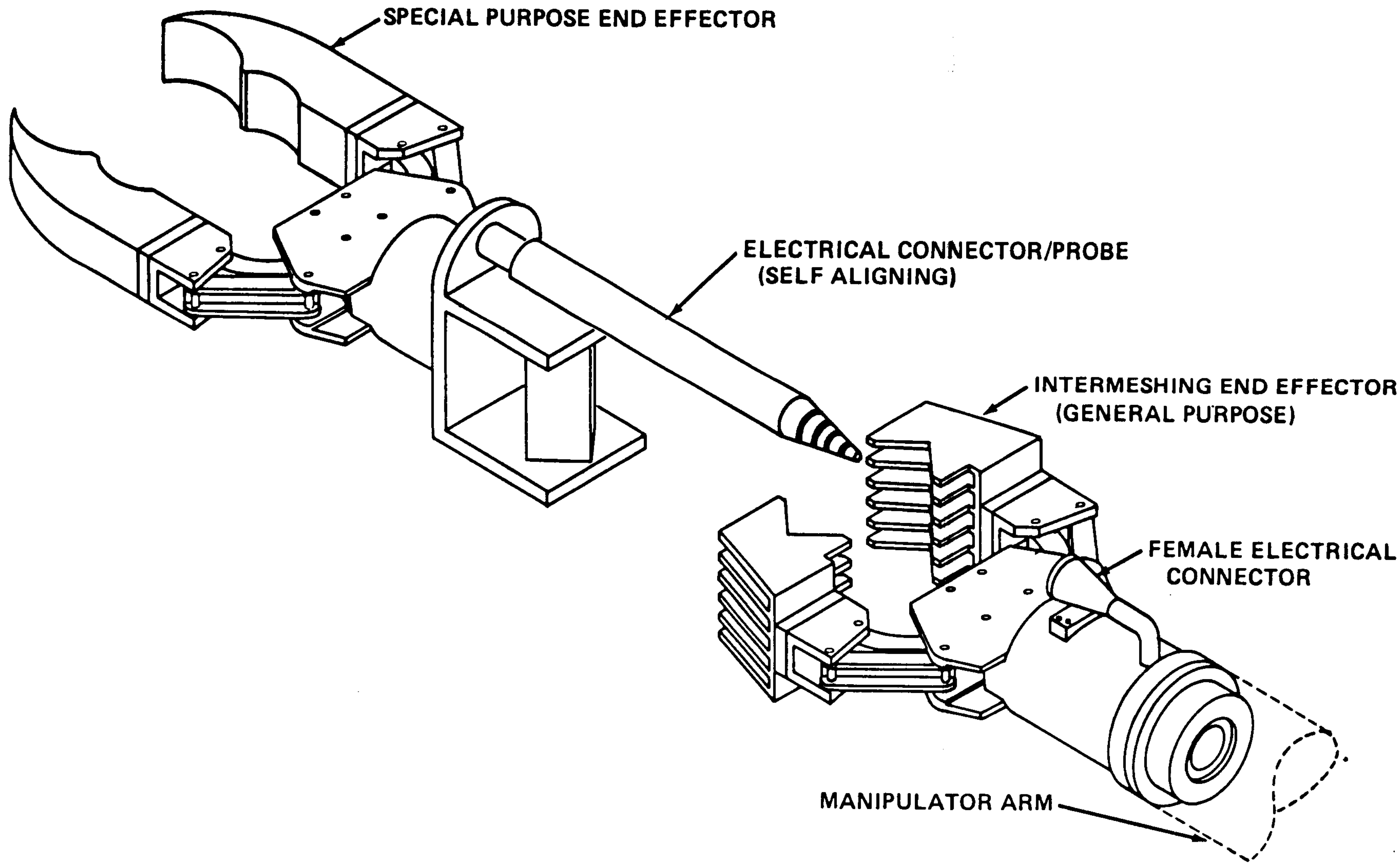
A functional diagram of the protoflight manipulator arm is on the opposing page. Depicted is a remote control station, software control algorithms, task board and the manipulator arm. Although the arm is hardwired to the digital computer and control station in the lab, in flight it would be commanded through an RF link.

DEXTEROUS SERVICING SYSTEM SIMULATION



REMOTE SATELLITE SERVICING

An intermeshing end effector was developed for the PFMA and so named because its fingers intermesh with each other like a tuning condenser when the hand is closed. The hand will grasp different size and shape objects. The intermeshing end effector will be utilized to evaluate and demonstrate a set of interchangeable tools. Shown on the opposing page is an intermeshing end effector ready to grasp a special purpose tool. A set of interchangeable tools could be stored and utilized by the intermeshing end effector on any given repair mission. Note also the capability of mating a conical electrical connector for a powered tool.

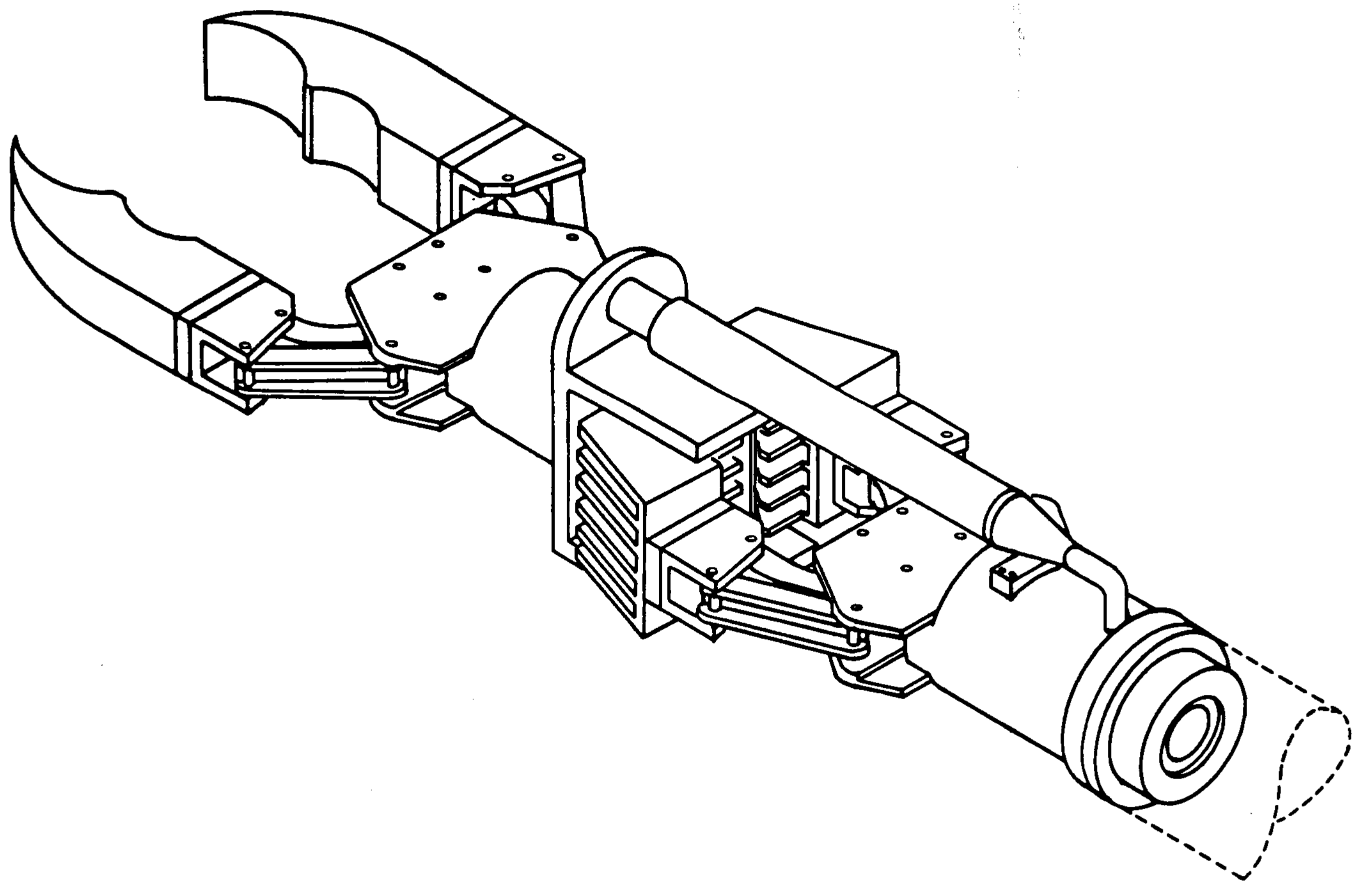


END EFFECTOR TOOL INTERFACE

6-0

REMOTE SATELLITE SERVICING

The opposing page depicts the intermeshing end effector grasping a specific tool. Testing to date indicates satisfactory results utilizing the intermeshing end effector as a general purpose grasping device in its ability to grasp different size and shape objects. Tools or objects designed for the end effector are easily and solidly grasped.



END EFFECTOR TOOL INTERFACE

REMOTE SATELLITE SERVICING

JPL is now developing a "smart" hand with force and torque sensing and grip force, adapting the intermeshing concept, that will be evaluated on the PFMA. Force and torque information will be displayed to the operator by a color scaled bar chart on a video screen or by numeric readout. The smart hand concept is shown on the opposite page.

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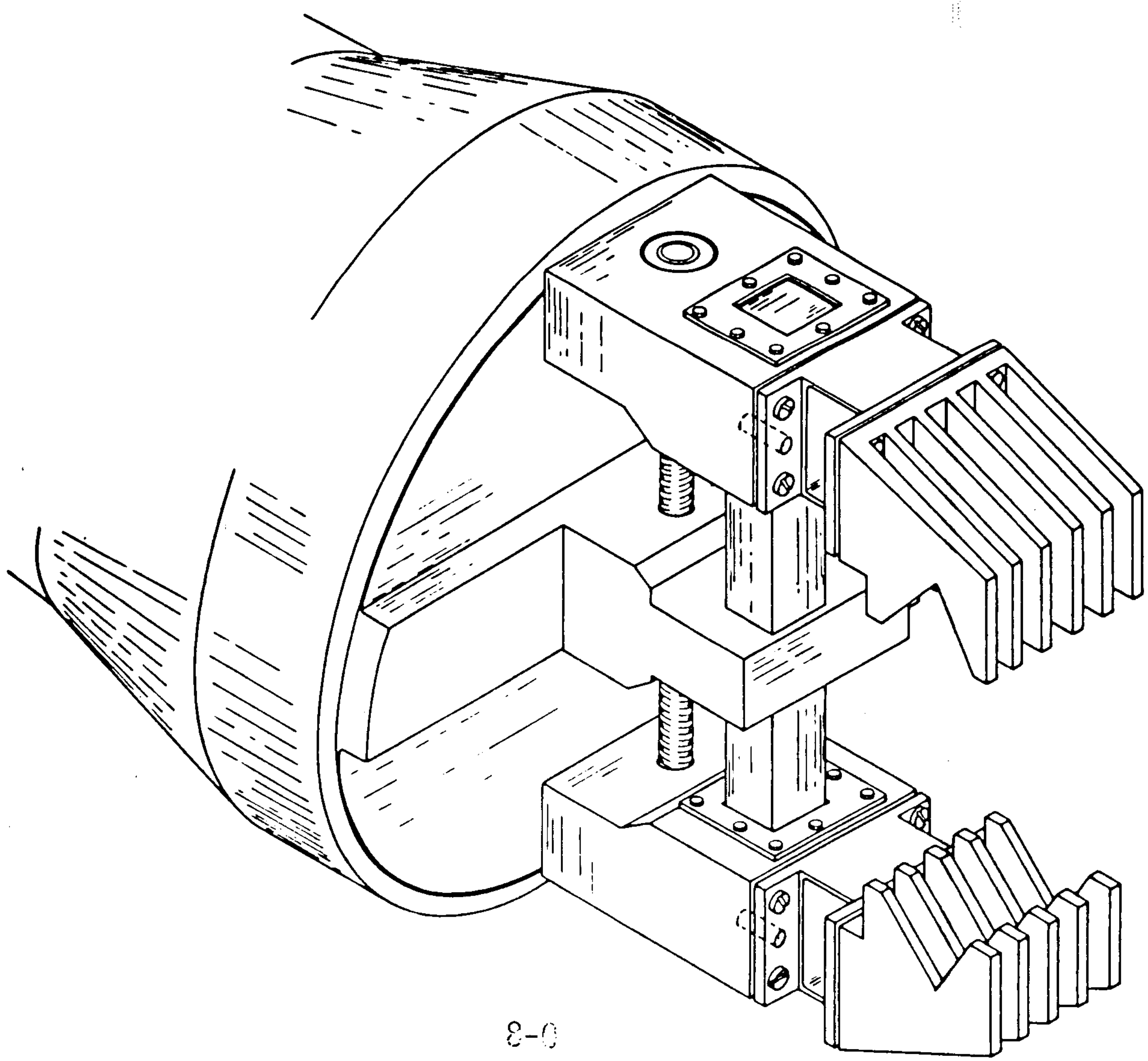
JPL

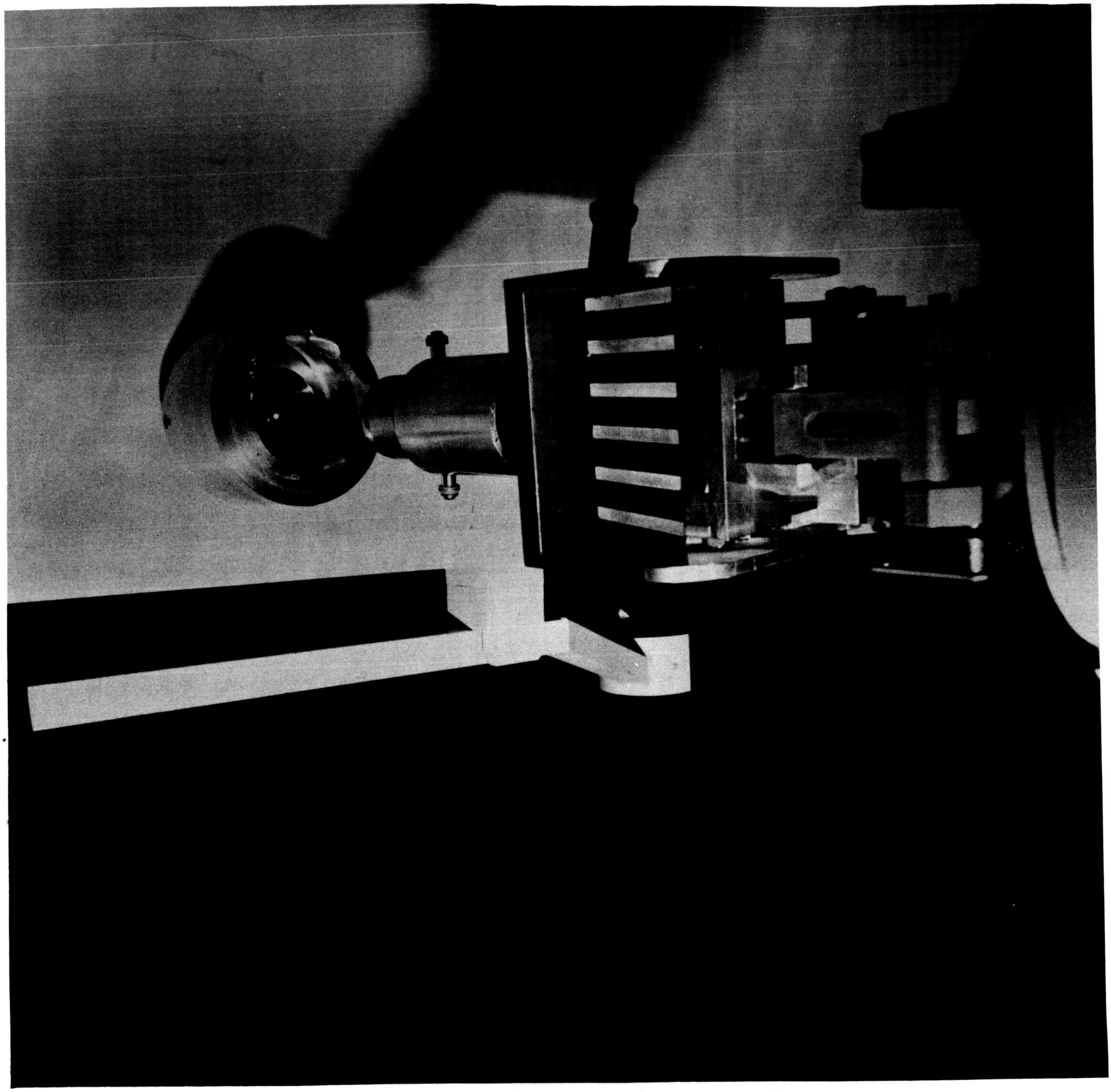
OVERALL VIEW OF SERVO GRIPPER ASSEMBLY WITH FORCE AND GRASP FORCE SENSOR AND INTERMESHING CLAWS

WITH ONE SQUARE SUPPORT COLUMN AND LINEAR ROLLER BEARINGS



8-0

The Fairchild fluid coupling sealed on both male and female sides when demated and open on both sides when coupled is shown here being mated by the remotely operated PFMA. The mating demonstrates umbilical fluid coupling by taking the male portion with umbilical attachment and coupling it to the female side mounted on a simulated bulkhead. Compressed air is transferred through the coupling to demonstrate a mated condition. To implement mating by the PFMA, the Fairchild coupling was modified by adding a cylindrical adapter to the female side, guides and a grasping handle to the male side. The functional portion of the coupling was not changed.



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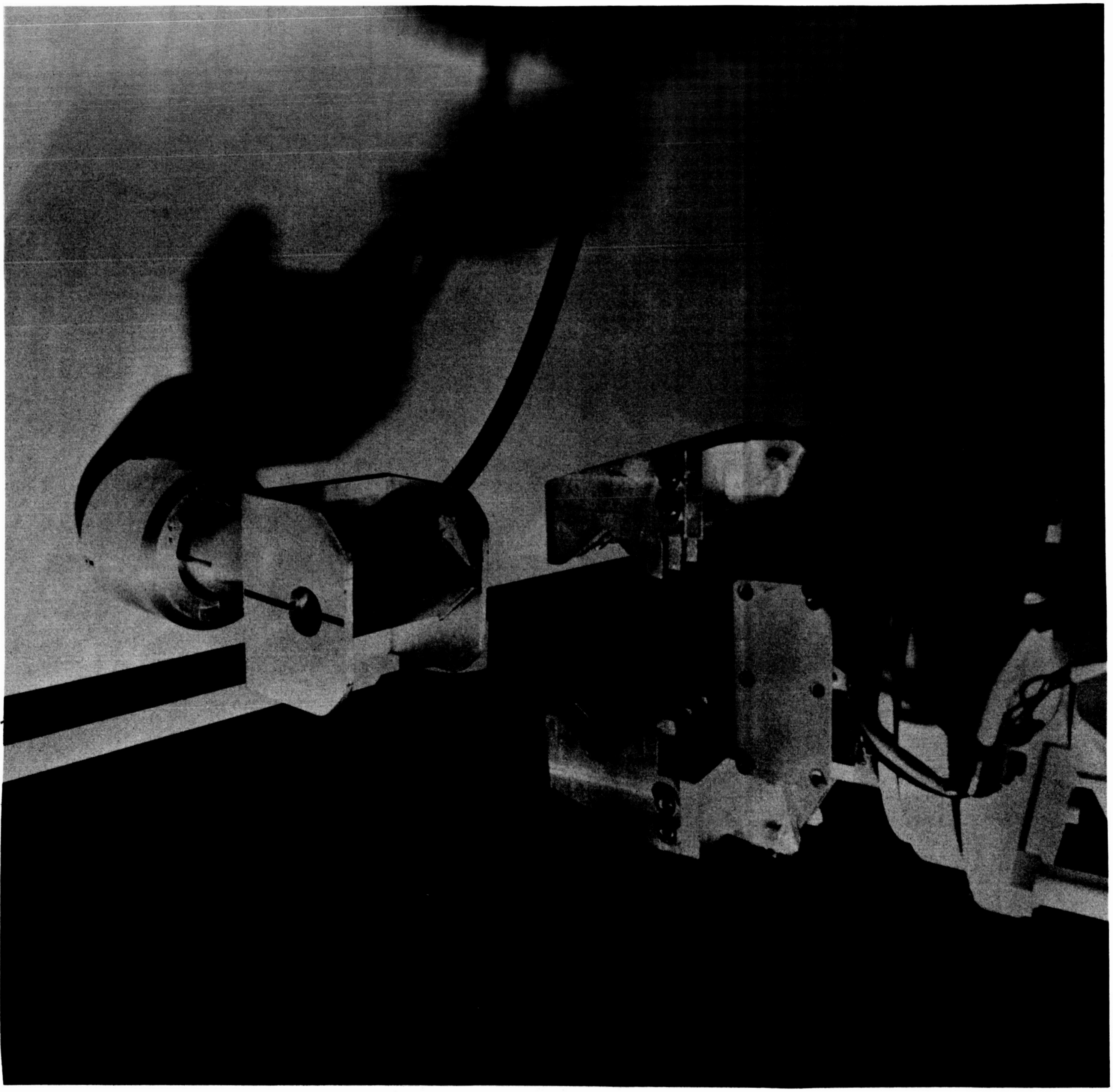
REMOTE SATELLITE SERVICING

The opposite page depicts the mated Fairchild coupling. The coupling/decoupling is achieved by a remote operator using a 6 DOF hand controller, end effector control mode (reference coordinate frame at end of arm), and two camera views of the action.

10-0



REMOTE SATELLITE SERVICING

Pictured below is the reconfigurable control station so named because it is designed for ease of changing to fit the operator or the situation. Compactness is less important since the baseline operation of remote manipulators will probably be from the ground. Shown on this control station are three TV monitors, two hand controllers, and a computer keyboard. Switches, meters, etc., could be mounted on the side panels as required. Insets and numeric overlays can be displayed on the TV monitors at the pilot's command or convenience.

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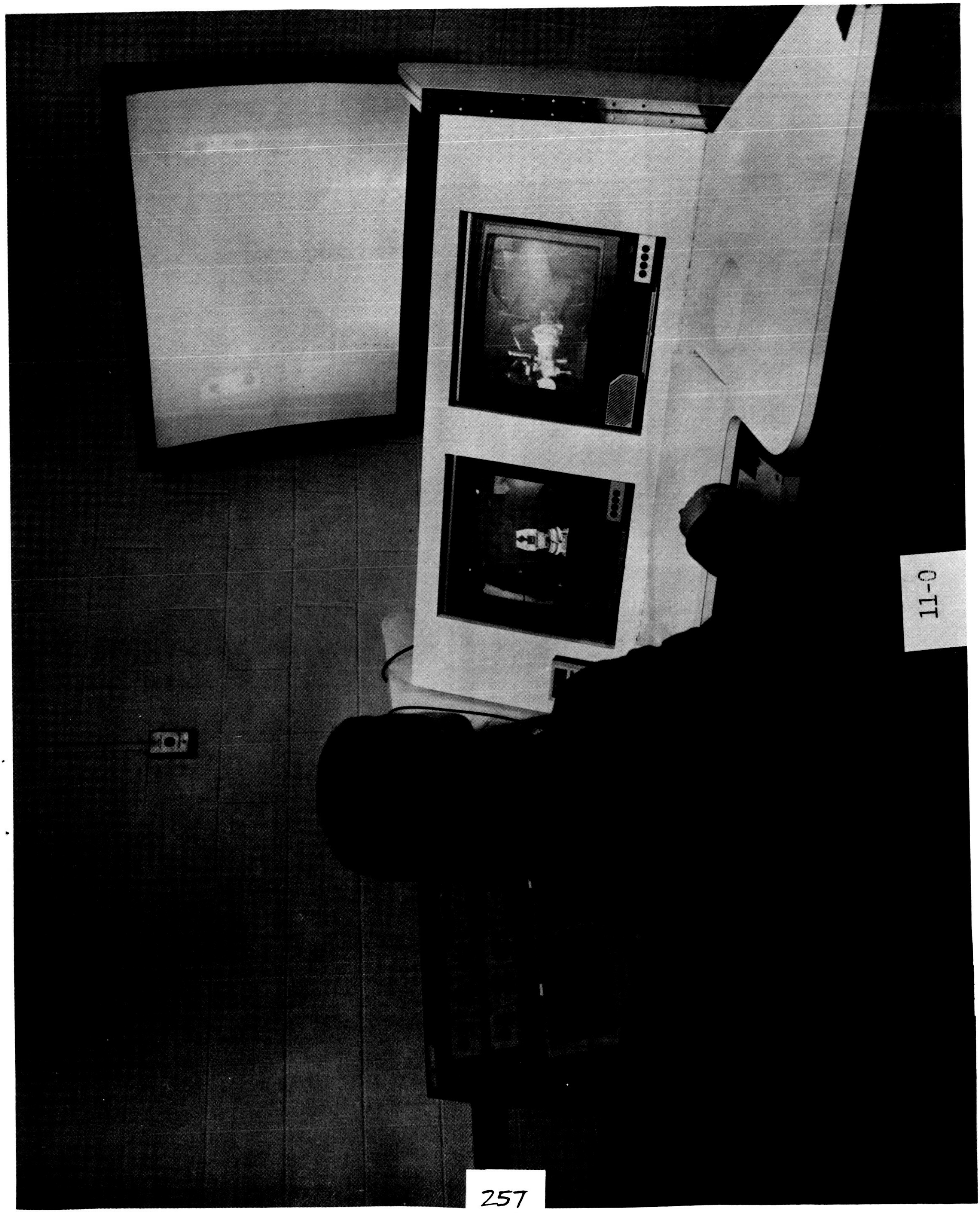

11-C

REMOTE SATELLITE SERVICING

The 6 DOF hand controller depicted here was developed by Canadian Aerospace Electronics (CAE). The single 6 DOF hand controller basically achieves three translational degrees of freedom from its base by X/Y/Z arm motion and three rotational degrees of freedom from the ball on top by orthagonal wrist motion. In addition to freeing one hand, a relative one-to-one correspondence between arm and wrist motion and the manipulator arm motion makes the single 6 DOF readily adaptable to the protoflight manipulator arm control.

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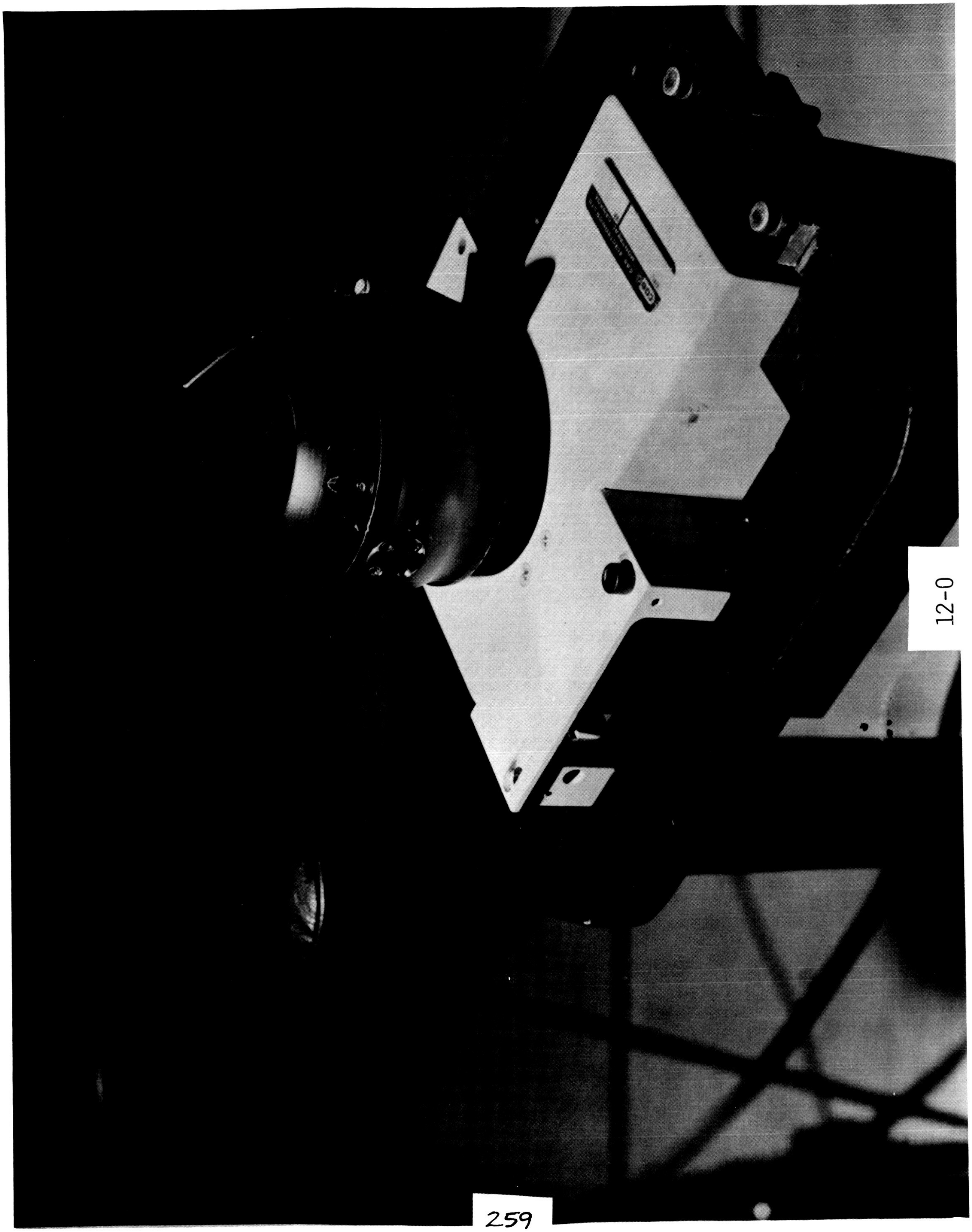
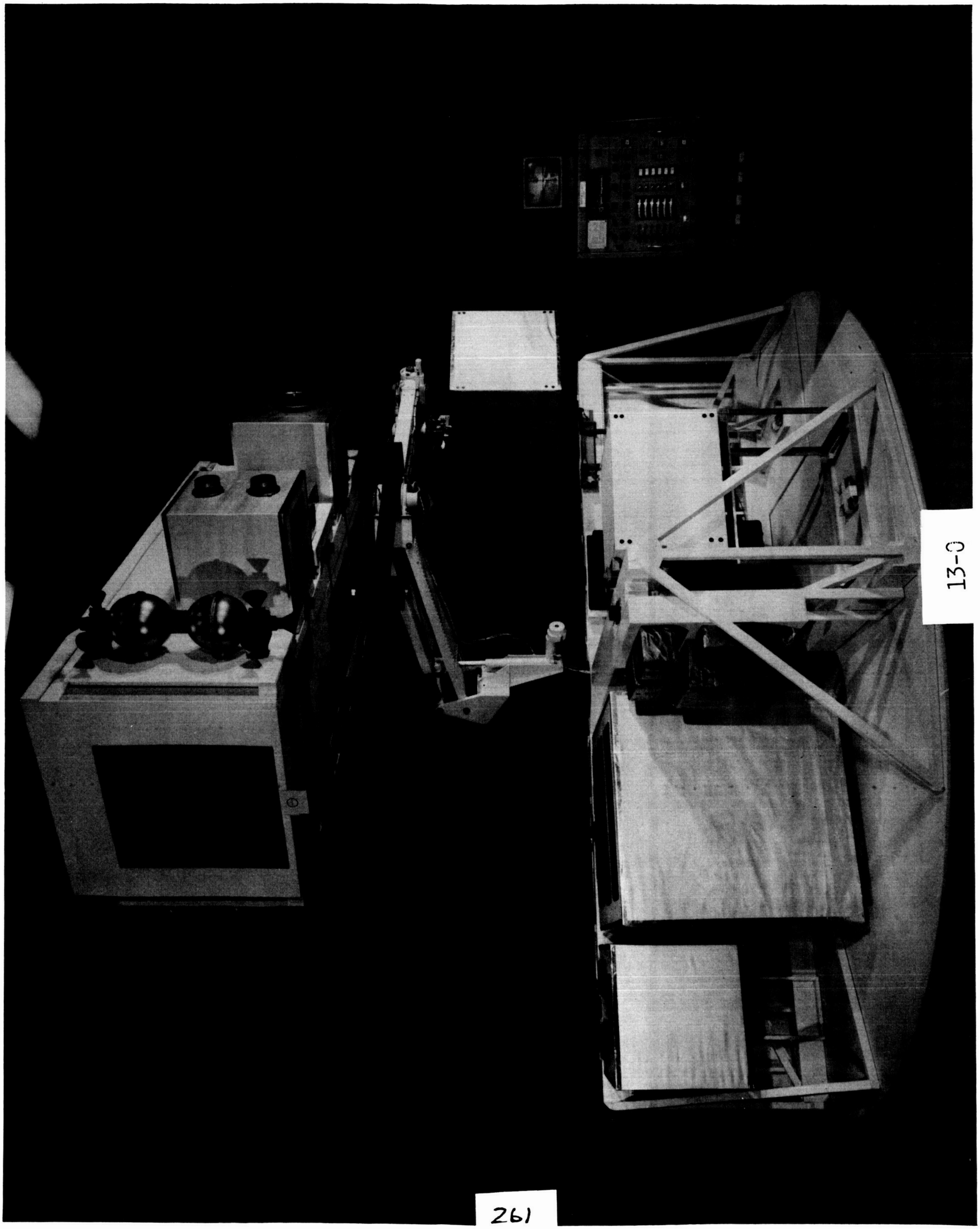


12-0

REMOTE SATELLITE SERVICING

INTEGRATED ORBITAL SERVICER SYSTEM (IOSS)

The orbital servicer system was designed specifically for satellite servicing. It is composed of a full-scale orbital servicer space vehicle mockup or storage rack, a satellite spacecraft mockup, a control panel, an interface mechanism and a 6 DOF mechanical manipulator arm. The 6 DOF manipulator arm and interface mechanism constitute the basis of the system. The storage rack and spacecraft are fixed relative to each other by a docking probe which simulates a hard dock of two vehicles. A 6 DOF manipulator arm is mounted on the center shaft (docking probe) that docks the two vehicles together. A remote control panel with potentiometers, meters, indicator lights, video monitor and manual drive switches is utilized to control the arm in the manual mode which is considered the backup mode of operation. A digital computer is used for automatic control and is the primary mode of operation. The function of the 6 DOF arm is to remove faulty or spent modules from the spacecraft and replace them with good modules from the storage rack on the orbital servicer by either manual or automatic control. Modules are attached to the interface mechanism.



13-0

REMOTE SATELLITE SERVICING

The Integrated Orbital Servicer System (IOSS) is depicted functionally on the opposing page. The operator essentially observes the automated module exchanges and interferes only in case of abnormalities. A manual and manual augmented mode of operation is provided as backup or redundant modes of operation. In real remote satellite servicing, an RF link would be between the servicer and the control station instead of the hard wire as shown here.

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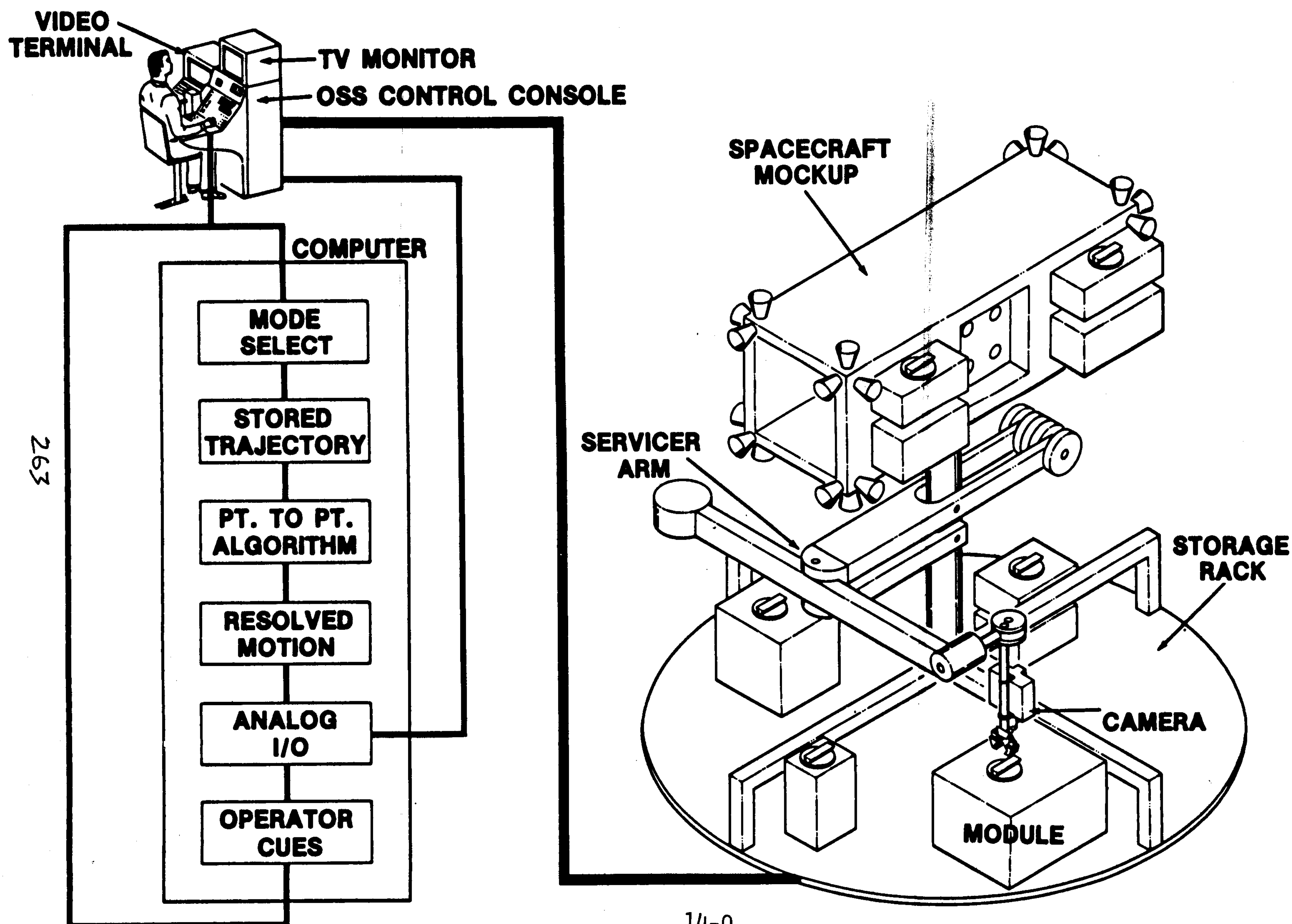
ORBITAL SERVICER SYSTEM



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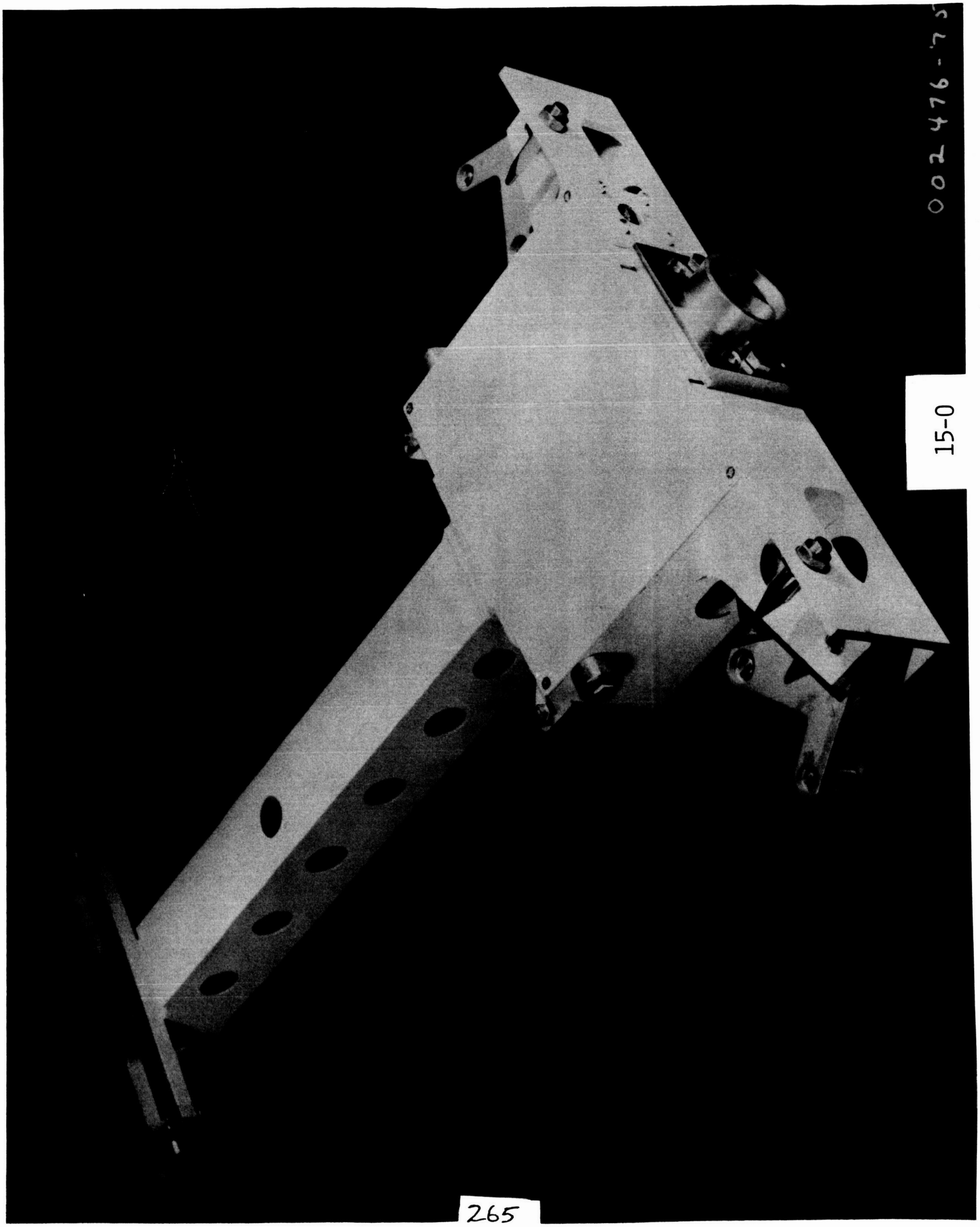
REMOTE SATELLITE SERVICING

IOSS SIDE-MOUNTED INTERFACE MECHANISM

The module interface mechanisms provide the structural attachment between the module and the spacecraft or the storage rack. It also provides the alignment and mating/demating forces for the connectors. The interface mechanism has two parts; a baseplate that is fastened to the module and a baseplate receptacle that is fastened to the spacecraft. The baseplate receptacle is passive. The baseplate has the linkages, cams, and rollers that latch the baseplate into the receptacle. The baseplate mechanism is mechanically driven from the servicer end effector. The interface mechanism is critical to automated modular servicing.

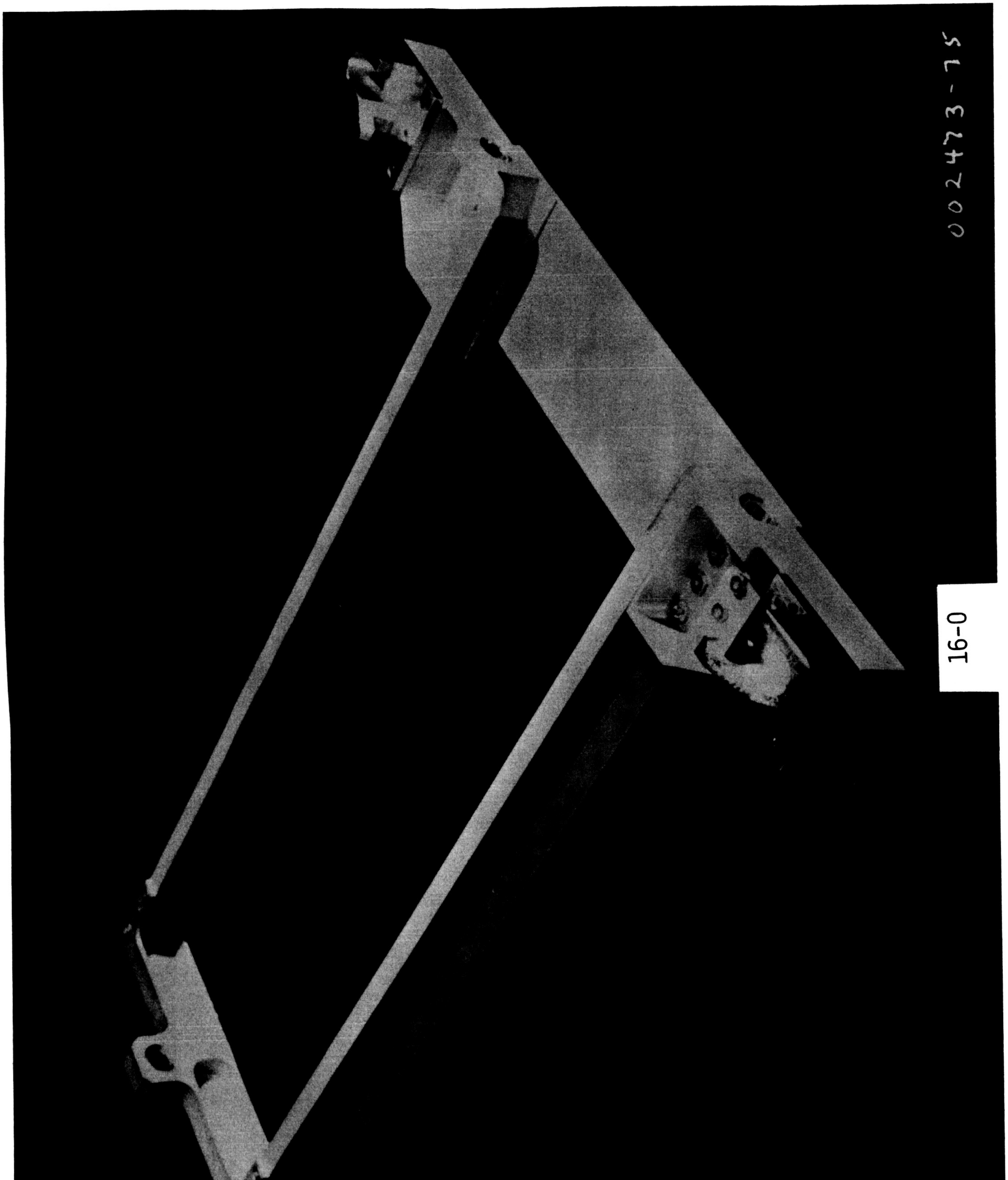
The IOSS is being modified to demonstrate remotely exchanging MMS modules utilizing the EVA adapter tool mounted on the end of the manipulator arm.



15-0

REMOTE SATELLITE SERVICING

The baseplate receptacle fastened to the spacecraft has track guides to receive the baseplate. Once the alignment tip on the baseplate is inserted into the guide track, it is forced down the guide track until completely mated. Electrical connectors not shown in these pictures are mated when the baseplate is fully inserted into its receptacle. The arm repeatability (accuracy) is about one-eighth inch which is well under the tolerances built in the hardware.



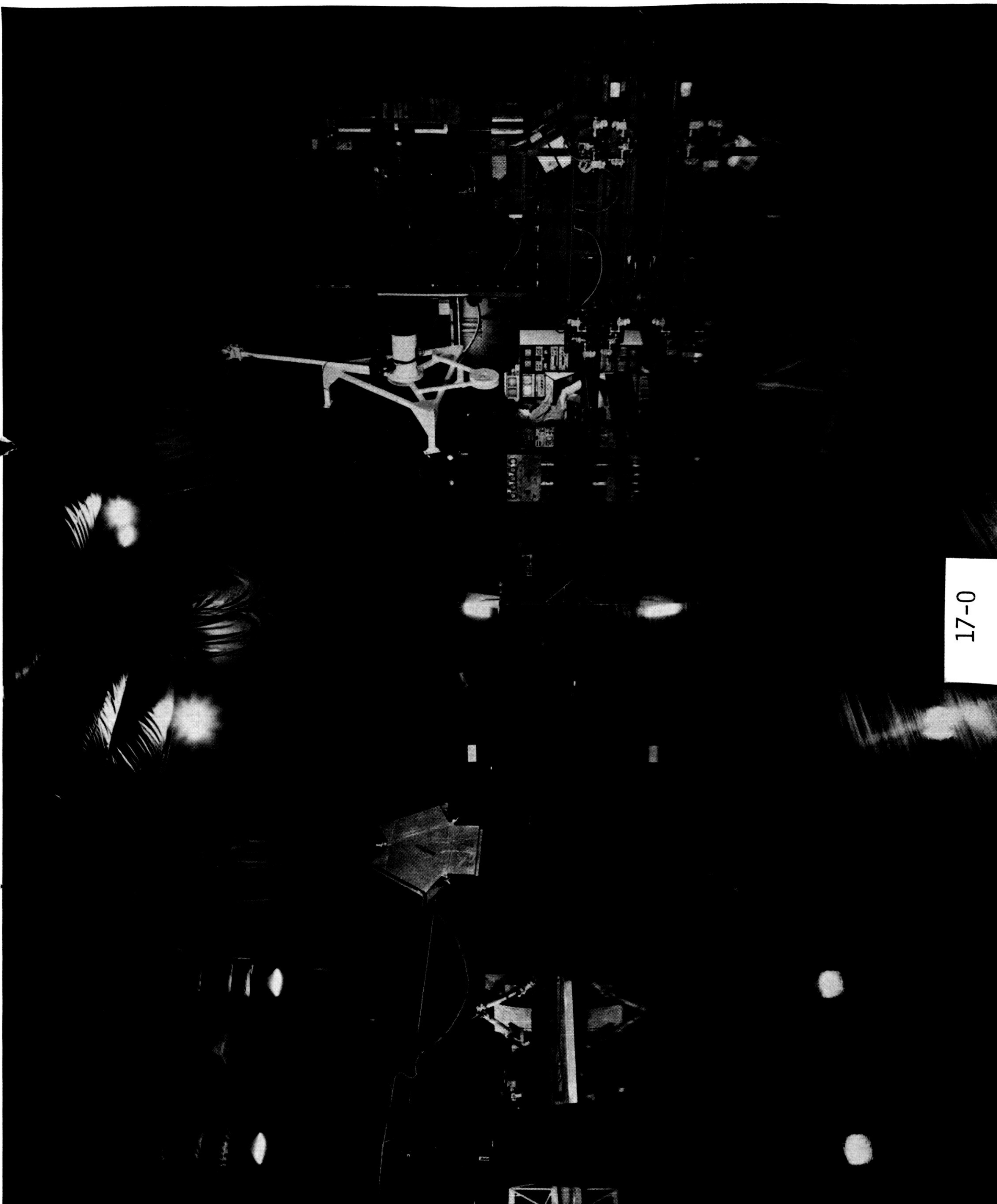
16-0

REMOTE SATELLITE SERVICING

AIR-BEARING FLAT FLOOR FACILITY

This facility consists of the following elements: (1) a 4000-square foot epoxy-coated surface; (2) a mobility vehicle (MV) floated on three air-bearing pads (providing three DOF) and electrical motor servo drives which provide the other three degrees of freedom; (3) a 6 DOF target motion arm at the edge of the epoxy surface, which provides complex motion (roll, coning, tumbling) to a suitable docking/capture target such as a Space Telescope mockup; (4) a reconfigurable pilot console; (5) RF links to/from the MV for video, commands, and telemetry; (6) video processing and display systems for simulation of special effects such as frame rate reduction; (7) computation facilities and reconfigurable software, both on the MV and off-floor interfacing with the pilot console.

For a given simulation problem, hardware and software in the appropriate combination are configured to provide a realistic representation of the actual space operation, e.g., a man-in-the-loop remotely-controlled docking between OMV and a target vehicle such as Space Telescope.



17-0

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REMOTE SATELLITE SERVICING

SUMMARY:

The focus of remote satellite servicing at MSFC is system oriented. The approach emphasizes ground-based experimental and simulation techniques involving autonomous as well as man-in-the-loop operations. Simulations include manipulator arms performing modular exchanges, more dexterous tasks and approach/docking operations utilizing an air-bearing 6 DOF mobility unit maneuvered on a flat floor. The total MSFC program involves a blend of system and subsystem investigations oriented toward remote operations to support teleoperation capabilities for future space missions.

There is little doubt that remote satellite servicing is practical with present-day technology, however, there must be a degree of complexity in which servicing would not be considered feasible. There are numerous key issues in remote satellite servicing that should be considered in the process of developing remote servicing ability. Some are listed on the opposing page.

Appropriate tasks should be determined and performed in simulations and ground demonstrations, ultimately leading to space flight experiment demonstrations. Further, experimental and simulation techniques for satellite servicing should be pursued to establish the degree of complexity in which servicing is practical.

REMOTE SATELLITE SERVICING

KEY ISSUES IN REMOTE SATELLITE SERVICING

- MANIPULATOR ARM/INTERFACE MECHANISMS/END EFFECTORS
- COMMUNICATION TIME DELAY
- COMMUNICATION BANDWIDTH LIMITATIONS
- VISION RELATED ISSUES
- TARGET LIGHTING
- CONTROL COMMAND CHARACTERISTICS
- INFORMATION PRESENTATION/DISPLAY
- SENSORS
- TOOLS
- INTERFACE MECHANISMS

To be presented at the Satellite Services Workshop II
NASA Johnson Space Center, Nov. 6-8, 1985

Concept for a Liquid Helium Servicing Kit

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INTRODUCTION

Storage and transfer of helium on the ground is a standard laboratory procedure over a wide range of volumes and flow rates. On the ground gravity plays a key role in separating the liquid and vapor phases and warm helium gas over pressure is used to develop the appropriate flows. In the Shuttle's low level random g environment, this phase separation technique cannot be utilized so that a new technique for separating and controlling the liquid vapor interface is required. A different approach to developing the necessary driving pressure head is also required. Gaging of the flow is typically done in the laboratory by monitoring liquid level. This is also dependent on controlling the liquid-vapor interface and thus a different approach is required in space. NASA has recognized the need for a helium transfer capability and is in the process of developing the necessary technology and systems.

USERS/NOMINAL MISSION

NASA's need for a helium resupply capability in space is being driven primarily by astrophysics missions. Table 1 is a list of the NASA missions which will require liquid helium servicing. In addition the DOD has identified several missions which require large quantities of helium. The Space Infrared Telescope Facility (SIRTF) and the Advanced X-Ray Astrophysics Facility (AXAF) missions would be the first NASA payloads requiring resupply. Since both of these observatories are baselined for extended lifetimes of approximately 10 years servicing will be required at two to three year intervals. If the space station develops into a true microgravity laboratory, then we would expect the helium supply needs to be equivalent to current ground based laboratory needs, which is on the order of several hundred liters per week. The need for such a liquid helium resupply capability was demonstrated dramatically by the 1983 Infrared Astronomical Satellite (IRAS) mission, which represented the first successful long term storage of helium for cooling of a space telescope. The mission lasted 10 months and the ability to continue the mission by refilling the 500 liter helium tank would have been eagerly accepted by the science community.

In order to develop a concept for a resupply kit or airborne support equipment (ASE), SIRTF was chosen as the payload. Since SIRTF represents one of the larger and nearer term astrophysics missions requiring resupply, an ASE capable of resupplying it would suit many other missions. The SIRTF cryogenic system is a nominal 4000 liter toroidal tank of superfluid helium. Figure 1 is a representation of a typical SIRTF servicing mission from either the STS or space station. There is a 1 degree per day difference in orbital precession rate between the STS and the nominal SIRTF orbit. The normal 2-4 day cooldown and fill operation combined with the limited OMV plane change capability ruled out the option of in-sutu servicing at this time.

The concepts presented here are a combination of work performed at Ames Research Center (ARC) and at Lockheed Missiles and Space Company (LMSC) and Ball Aerospace Division (BASD) under contract to ARC.

HELIUM PROPERTIES

Open cycle evaporative cooling of helium results in the lowest temperature that can be obtained from any cryogens. The temperatures reached by helium are critical for long wavelength infrared detectors as well as for exploiting superconducting phenomena. Helium is a quantum fluid and exhibits several strange properties. It is phenomenologically represented by a two fluid model representing a superfluid and a normal fluid component in which the superfluid component carries no entropy. The transition from normal to superfluid behavior occurs at reduced temperatures (2.12 Kelvin) and this phase transition is characterized by a rapid reduction in viscosity and increased in thermal conductivity. A unique characteristic of the superfluid state is that it can move freely in small geometries while the motion of normal fluid is hampered by viscosity effects. In the presence of heat loads this results in a pressure head which can be used to control the liquid-vapor interface. In addition, the extremely high conductivity of the fluid prevents stratification which is a severe problem in normal fluids. Finally at 1.8 Kelvin, the fluid is about 15% denser than at atmospheric pressure which results in more efficient packaging, higher effective cooling power per unit volume, and a larger latent heat in the boiloff vapor. It is apparent that storage and use of helium in the superfluid state offers several distinct advantages over the normal fluid state. The decision was therefore made to baseline a superfluid ASE.

MAJOR ELEMENTS/INTERFACES

The three major elements of a servicing mission are shown in Figure 2. The external ASE kit includes the helium storage dewar, the transfer line and couplers, pressure, level, flow and temperature sensors and drive electronics for the valves and sensors. For the EVA hookup of SIRTF and the helium ASE, tools for the transfer line and electronic connectors are required. In addition, a high pressure helium gas bottle and "sniff" type leak detector are supplied to verify the vacuum integrity of the transfer line connections. The ASE command console can be located on the aft flight deck of the STS or on the Space Shuttle. It monitors and controls the transfer through the ASE controller and the SIRTF controller. In trying to build an ASE kit that can be used on multiple payloads, the control of the payload valves and instrumentation presents a major interface problem. The cooldown and fill process requires that the temperature of the receiver tank be monitored and the fill/vent valves controlled. Standardization of payload, electrical connectors, valves and instrumentation is not thought to be practical or technically advisable since the ASE will service payloads developed over a 10-20 year period and valves and instruments will be upgraded during this period. Therefore the interface must be provided through the payload controller. Since the fill may create temperature and pressure conditions outside of the payload's normal operating range, the ASE controller must have override authority over the nominal payload controller. The easiest way to accomplish this is through the STS or SS command/data bus.

SYSTEM SIZING

The typical dewars used for ground storage which are in the 1-2% mass loss per day class would be satisfactory for STS operations where the longest storage interval would probably not exceed 14 days, but not adequate for the space station where the resupply kit may be resident for 2 months to a year. In designing the ASE, the decision was made to use IRAS/COBE storage technology as a minimum which would result in .2-.3% per day boiloff.

The supply dewar is sized for the worst case of filling a warm (300K) SIRTF cryogen system. This capability would be required if a servicing opportunity was missed and the telescope ran out of helium or if the telescope is warmed deliverately to allow repair of normally cold components. Enough helium has to be available to remove the specific heat of not only the tank but also the SIRTF telescope and thermal insulation system. The quantities of helium required are critically dependent on the limiting thermal impedance between the helium coolant and the telescope. If there is a poor connection, then high flow rates only waste helium. Since the heat cannot be removed fast enough, the tank fills but large amounts of heat continue to flow into the tank from the uncooled instrument payload and subsequent topoff is required. If the flow is throttled to match the payload heat rate then the transfer line parasitics can become a significant inefficiency. SIRTF is taken as an example of one of the largest helium systems and the thermal conductance from its tank to instruments is relatively poor. The cooldown and fill from ambient temperature would take 10,000 liters of helium.

An alternative to an all helium system would be to precool the telescope with a cryogen that has a higher heat capacity. If liquid hydrogen is used to precool the system to 20 Kelvin, the helium quantity can be reduced dramatically. The amount of hydrogen required for precooling is also much smaller than the helium required for cooling from 200-20K. The price for this reduction is quite high. A hydrogen tank, manifold and transfer line will be required. In addition, the payload will require separate precooling loops to prevent contamination. The ground and flight handling as well as ASE flight qualification becomes more complicated when handling a dangerous gas such as hydrogen. The complications to both the payload and the ASE are not worth the advantage unless, as may be the case during the Space Station (SS) era, hydrogen is readily available.

ASE CONFIGURATION

A strawman layout of the external ASE is shown in Figure 3. It takes up about 1/2 of the STS bay diameter over a 3 meter length. The geometry of the tanks with the manifold on the X axis allows the fill vent lines to be situated in the top corner of the tank as shown which facilitates operation and top off in either the horizontal or launch orientation. The cylindrical shape allows for implementation of low heat leak support systems using current IRAS and COBE technology. In addition, the regular cylindrical geometry allows for easy liquid level gaging. The 3 meter overall length allows for storage of transfer lines on the dewar body. The dewar has a keel fitting and two sidewall trunion fittings to attach it to the Shuttle and the dewar outer shell vacuum vessel serves as the primary support structure for the external ASE. The tools, electronics, and manifold use the dewar as the support structure and are located on top (+Z) for easy accesss during ground and space operations. A small mechanical pump for maintenance of the superfluid helium pressure during ground hold is shown. Recent experiments on Space Lab 2 indicate that if the dewar is allowed to warm under its own leak and pass through the lambda transition to the normal fluid state, it could be pumped down in space with only a 10% penalty in cryogen if the bulk helium warms to 2.4K. The A prime cradle is shown located near the fill vent manifold. The cradle is used to hold SIRTF during servicing and the location minimizes the transfer line length.

INTERNAL CONFIGURATION

A strawman configuration for the details of the ASE is shown in Figure 4. The helium is controlled within the storage dewar using galleries to collect the liquid. This liquid is preferentially distributed on the tank walls due to the dominance of surface tension effects in zero g. The galleries collect the liquid and a sponge matrix at the inlet to the pump minimizes vapor entrainment except during the highest disturbances (thruster firings). The pump is a submerged centrifugal pump which utilizes an AC motor and impeller to drive the helium through the transfer line. A prototype of this pump has been demonstrated for use with both the superfluid and normal helium. During storage periods, the temperature in the storage dewar is maintained using a porous plug phase deparator and vapor cooled shields. During transfers the submerged pump dissipates approximately 10 Watts. To accommodate this large dissipation, a short low impedance vent line has been added to the storage tank. During a top off in which the SIRTF tank is already partially full, the bypass plumbing on the SIRTF manifold allows the transfer line to be cooled without blowing hot gas through the tank.

SYSTEM CHARACTERISTICS

The size of the system and critical interfaces are shown in Table 2. The overall mass is driven by the 10,000 liter storage tank. The standby power requirements are minimal and the operational power is well within the STS and SS capabilities.

OPERATION

The best case for a SIRTF topoff is when the system is still cold. The payload must be retrieved and brought into the vicinity of the STS cradle. An EVA is used to connect the payload and the ASE together mechanically and electrically, and also to connect the payload to its aft deck controller. Electrical, mechanical and vacuum integrity are verified during the EVA. Any payload unique servicing equipment or protective covers are installed during this maneuver. After being configured for the helium transfer, the helium aft flight deck ASE commands both the kit and the payload into the cooldown configuration. After the transfer line cooldown operation, the valve configuration is then changed to begin collecting liquid and the controller sets the flow rate predetermined for the payload. The entire top off operation of a 4000 liter system takes approximately 8 hours as shown in Figure 5a.

The worst case transfer occurs when the payload is warm. Although the system could be precooled using mechanical refrigeration or passive radiators, the cooldown from 300 Kelvin is taken as the worst case fill. The initial operations of the transfer are identical to the topoff case but the transfer line does not have to be cooled down using the bypass. The rate depends critically on the thermal link of SIRTF to the helium. Two ground based examples are the IRAS and COBE cooldowns. In the first case, 48 hours were required because the telescope was weakly coupled to the helium. For the COBE cooldown, due to an improved thermal link, the time was cut in half to 20 hours. If we scale these times to the larger SIRTF system, two to three days will be required. The total time including EVA's and redeployment will take approximately 5 days as shown in Figure 5b.

DEVELOPMENT SCHEDULE AND TECHNOLOGY

Schedule for development of the ASE is dependent on the first payload requiring helium transfer and its development schedule. Interface definition (both electrical and mechanical) are required. For payloads which are planned for launch two to three years ahead of the helium transfer kit, this presents a problem in the normal hardware development timeline. Specifically the bayonet connectors for the transfer line need to be defined. Table 3 is a list of all the important technologies and an assessment of their status.

CONCLUSIONS

Servicing of most large liquid helium systems will require retrieval and operations at the station or the STS. The ASE itself will be large and most efficient and easiest to handle if it stores helium in its superfluid state at reduced pressure. The couplers for the transfer line are a pacing item since retrofitting them on already launched payloads will be difficult. The requirement for cooldown in space almost doubles the quantity required for larger payloads. The ASE should be designed with aerospace long life dewar technology to allow operation on either the STS or SS. Several critical technologies, especially the pumps and gaging, are not well understood and need further ground development.

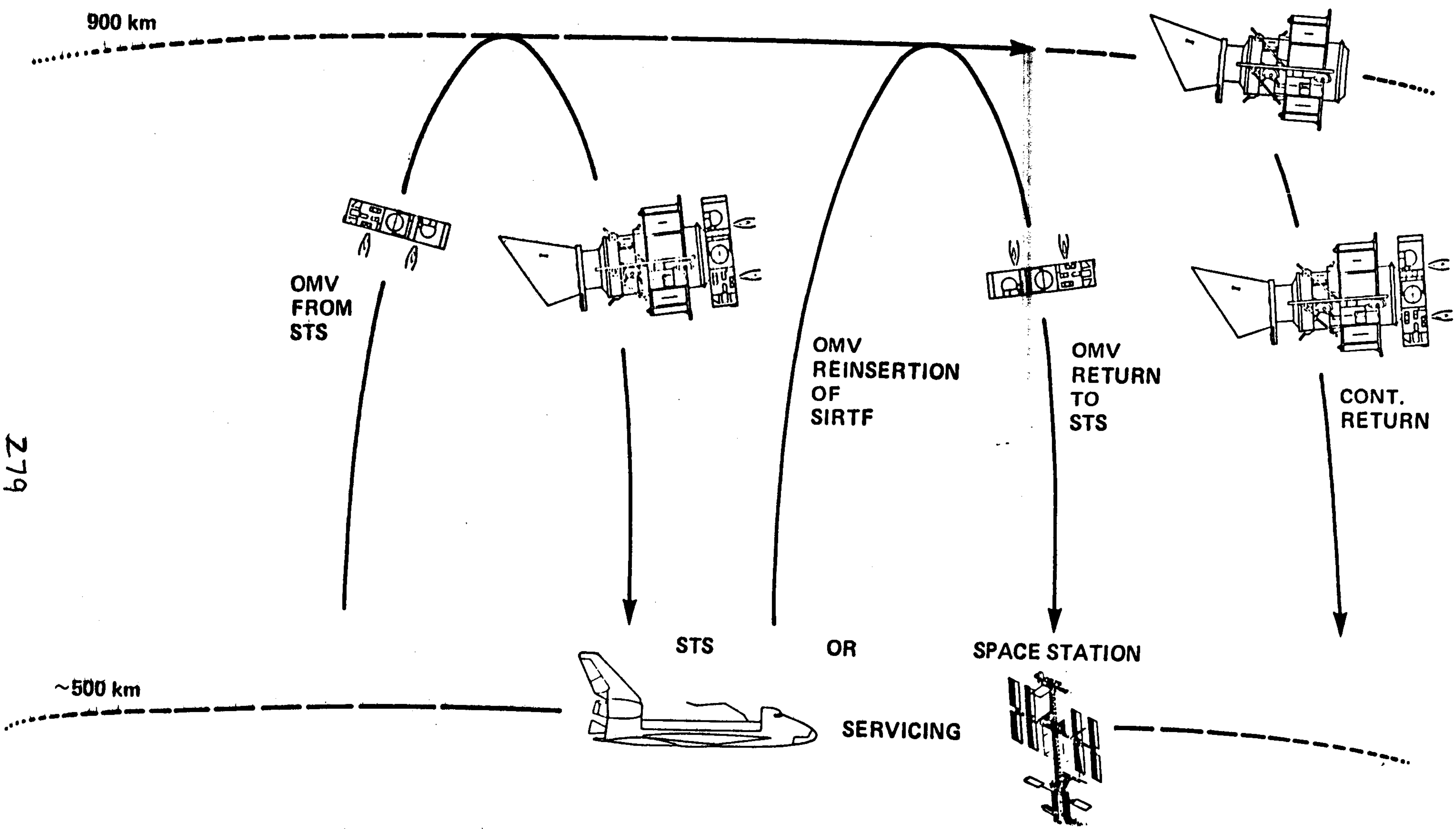
TABLE 1. POTENTIAL LIQUID HELIUM TRANSFER USERS

MISSION	OBJECTIVE	SERVICING LOCATION	SERVICING INTERVAL	SIZE (LITERS)	FIRST SERVICING DATE
SIRTF	IR ASTR[1]	STS/SS	2.0 YR	4000	1996
AXAF	X-RAY ASTR[1]	STS/SS	1.6 YR	500	1996
LDR	IR ASTR[1]	SS	2.0 YR	7000	1999
SUPERMAG	CHG/MASS	SS	2.0 YR	--	1996
GP-B	RELATIVITY	STS	2.0 YR	1600	1996
DARK SKY	SPACELAB 2 IRT REFLIGHT	STS PAYLOAD	7 DAY	200	--
ST	ASTR[1]	STS	3.0 YR	2000	1994

[1] ASTR = ASTROPHYSICS

TABLE 2. AIRBORNE SUPPORT EQUIPMENT PROPERTIES

MASS (FULL)	5,000 KG MAXIMUM
HELIUM VOLUME	12,000 LITER MAXIMUM
POWER CONSUMPTION	
EXTERNAL ASE KIT	<2W
STANDBY	<2W
TRANSFER	<200W
INTERNAL ASE KIT	<2W DURING STORAGE (<100W MAXIMUM)
STANDBY	<2W
TRANSFER	<100W
OUTER SHELL TEMPERATURE	300K $\pm$ 50K
MECHANICAL INTERFACE	SIRTF KEEL/TRUNION FITTINGS
RMS/MRMS INTERFACE	NASA GRAPPLE FIXTURE



SIRTF RETRIEVAL, SERVICING, REBOOST, CONTINGENCY RETURN

Fig. 1 OMV/SIRTF MISSION SCENARIO

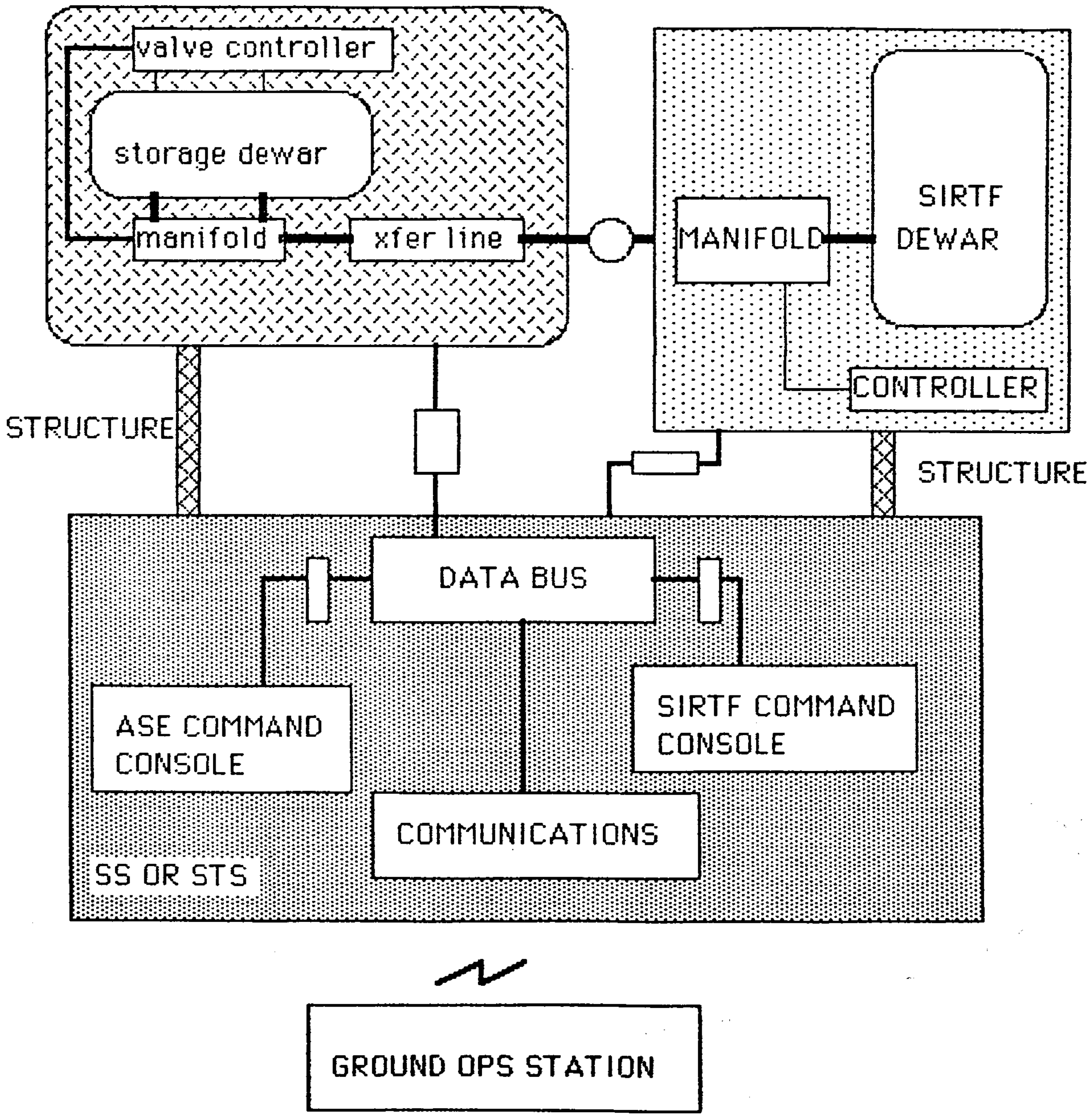


Fig. 2 Major elements of ASE on space station or Shuttle

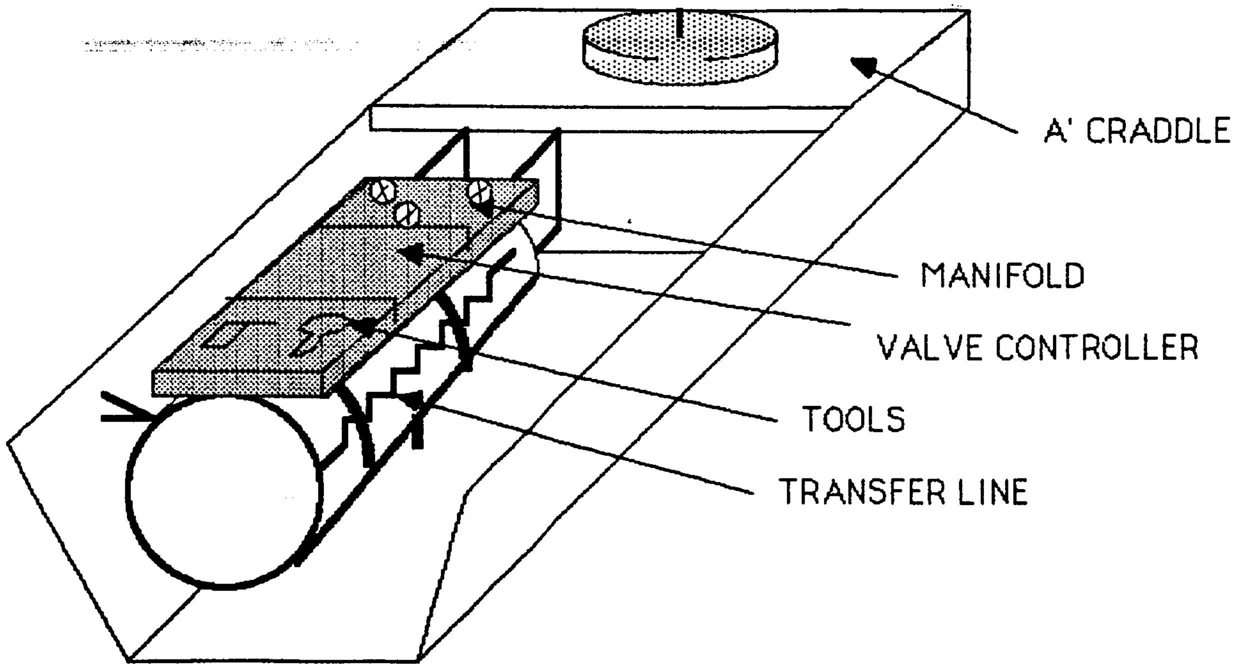


Fig. 3 Schematic of Servicing Kit in STS bay
length 3 meters and diameter 2 meters

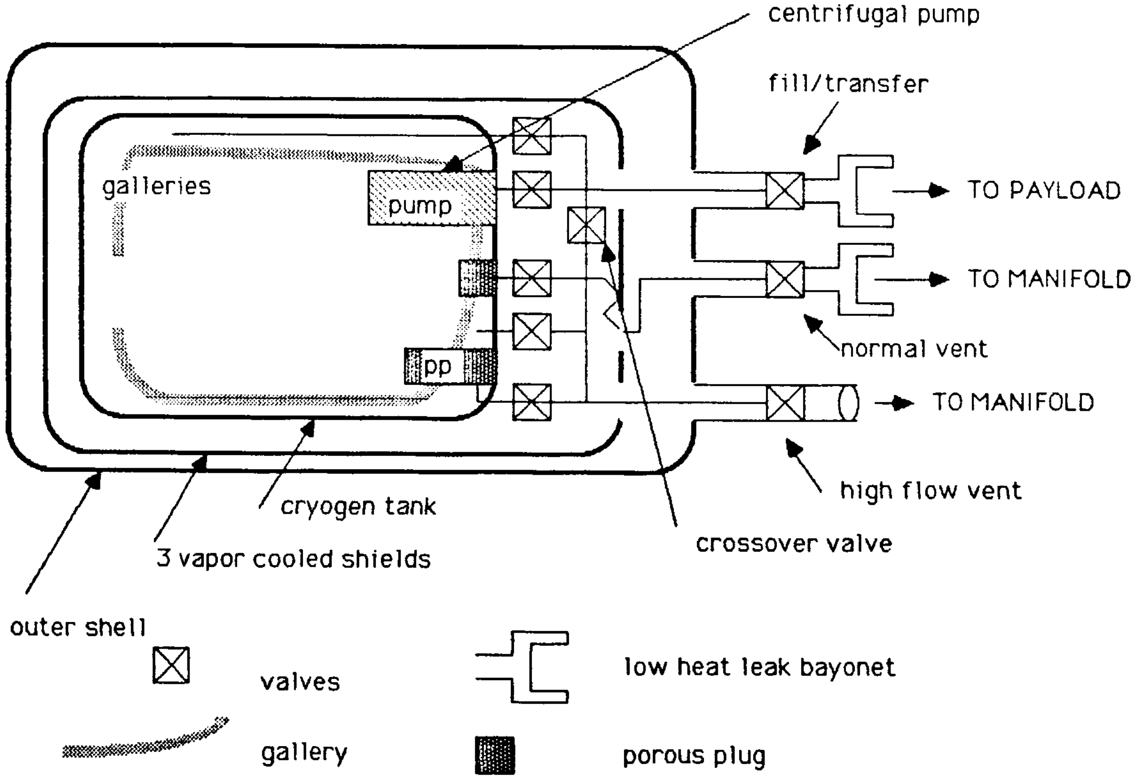


Fig. 4 ASE storage tank internal design

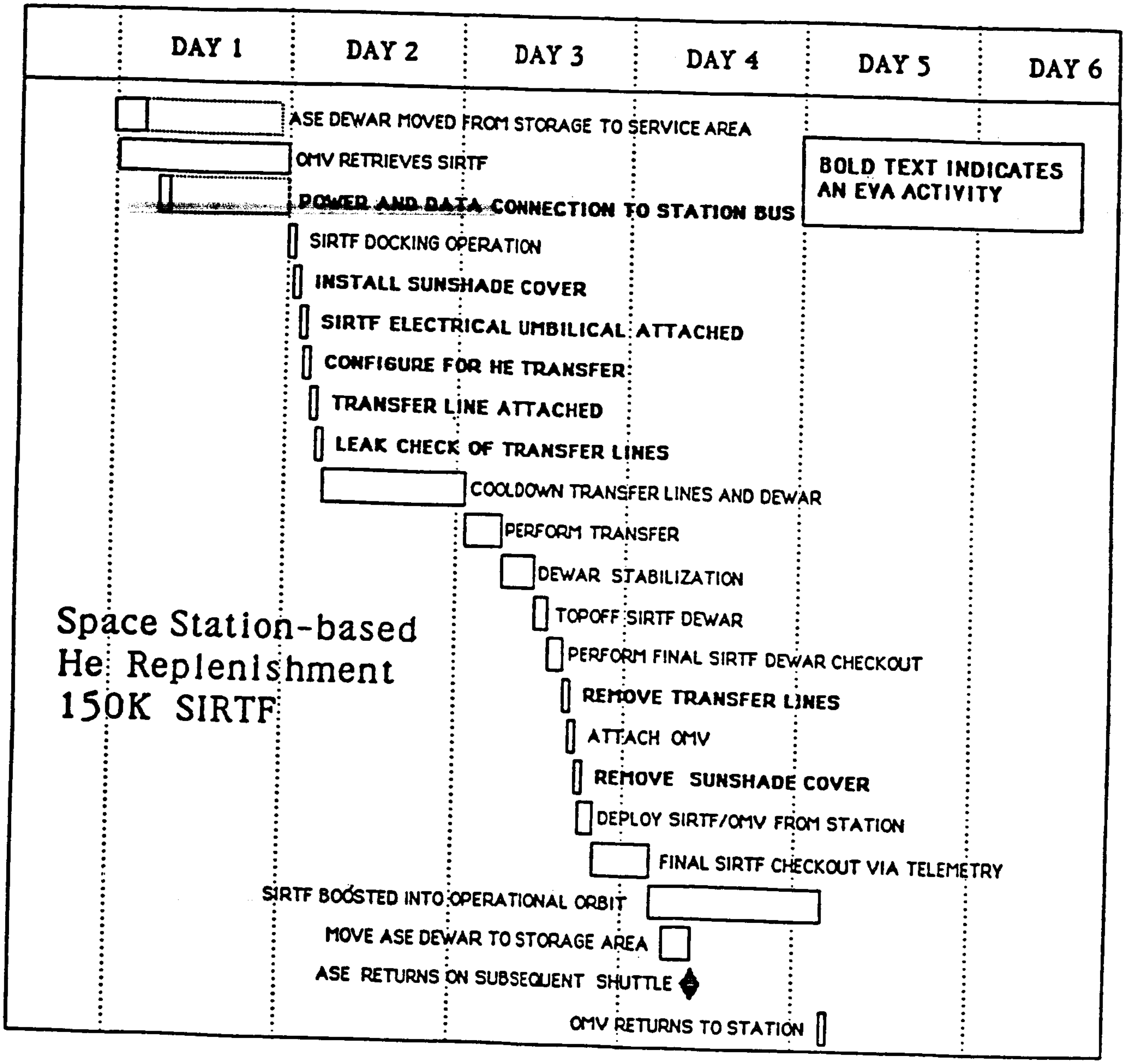


Fig. 5 Timeline for Station-Based Replenishment of Warm SIRTF

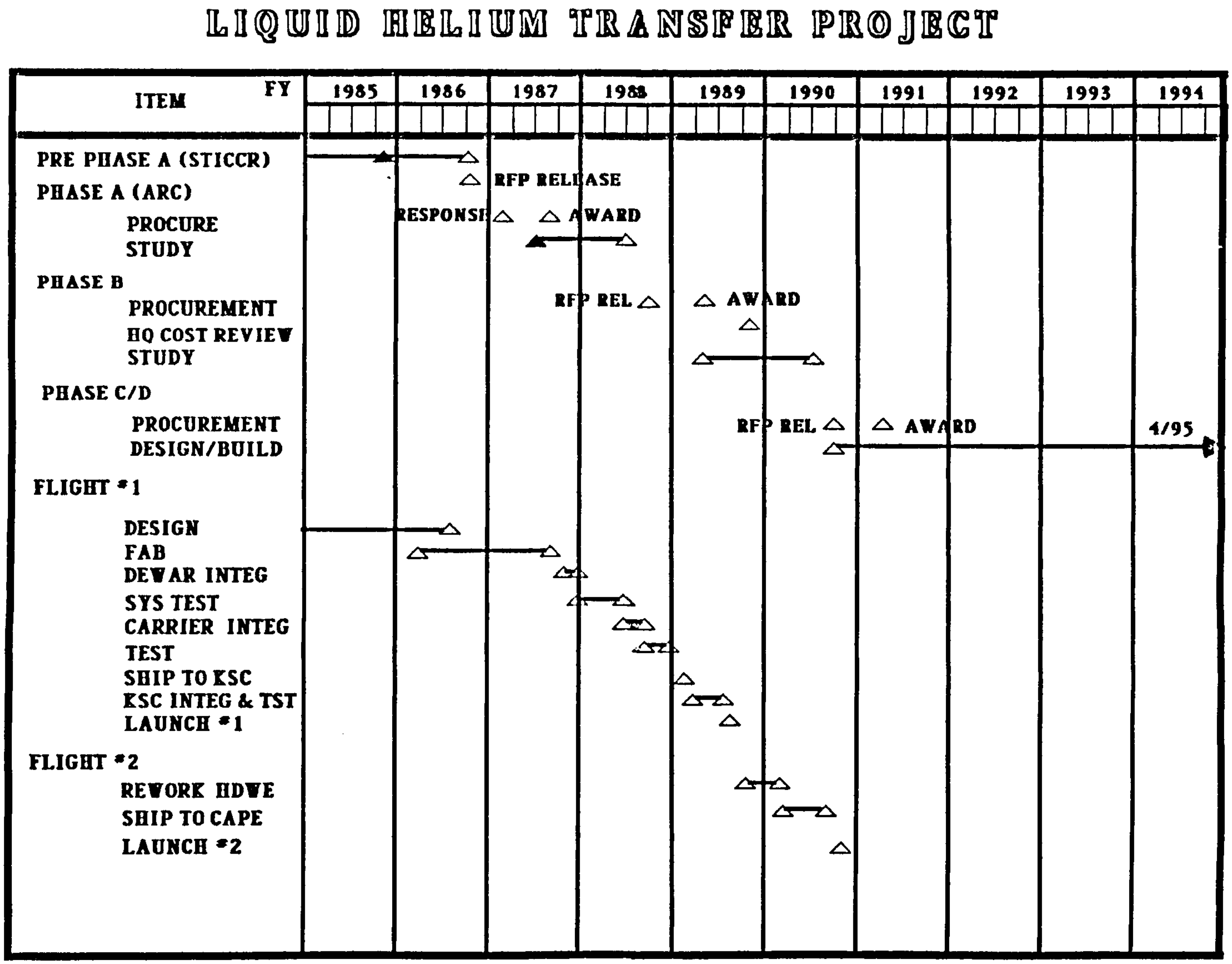


Fig. 6 Schedule for development of Liquid He Servicing Kit

SATELLITE SERVICES WORKSHOP II

POLAR ORBIT PLATFORM SERVICING

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POLAR ORBIT PLATFORM SERVICING

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ABSTRACT

Polar orbiting platforms offer unique opportunities to conduct full global observational experiments over extended periods, thus gaining new knowledge of the Earth's environmental trends and changes. During the life of the platform a key factor in the success of operations will be the maintenance, repair, and replacement of instruments and other platform elements which are life-limited. This will be achieved by servicing the platform in orbit.

The potential for extended use of orbiting platforms is greatly enhanced with the capability for service. In the past, when components failed, the instrument or platform was lost unless it had redundant components (added mass). In the next decade it should be possible to have near continuous platform functioning and to rely more on the capabilities of on-orbit servicing, including the ability to upgrade or enhance instruments or components on the platform if new technology becomes available after the platform is initially placed in orbit. This paper explores some realistic servicing scenarios based on current known National Space Transportation System (NSTS or STS) Orbiter servicing capabilities and representative platform instrument configurations and servicing requirements. Some assumptions are made about STS enhancements, and attention is also given to the probability that the Orbital Maneuvering Vehicle (OMV) will become an operational vehicle as part of the Space Station Program, and in the platform placement era (1993-1998) will have the potential for polar platform servicing and/or retrieval.

INTRODUCTION

This paper will consider four separate aspects of the problems of servicing in polar orbit:

I. Platform and the servicing vehicle rendezvous scenarios: Advantages and disadvantages for 1) in-situ servicing by the Orbital Maneuvering Vehicle (OMV); 2) transfer of the platform via OMV retrieval for STS servicing; or, 3) transfer via platform integral propulsion to STS altitudes for STS servicing.

II. Generic and specific servicing requirements and guidelines for the various servicing scenarios

III. Platform configurations being considered and compatibility with servicing vehicles

IV. Assumptions and issues for further study

Initially, concepts for large platforms with approximate mass of 35,000 kg were considered for the polar orbits.1 However, problems of multiple STS launches, assembly on orbit, and transfer of large masses to the higher altitudes of scientific interest have made the large platform less attractive. Several smaller platforms also have the advantage of being able to study the environ ment from different altitudes simultaneously. They are easier to launch and easier to service.

Currently four platforms of about 9,000 to 12,000 kg are planned for polar orbit operations beginning with the launch of the first in 1993. The three altitudes of major interest to the primary platform users** (Earth Observ ing Systems-EOS and the National Oceanic and Atmospheric Administration-NOAA), are sun-synchronous orbits at 824 km for the first and third platforms, 705 km for the second platform, and 540 km for the fourth platform.2 The servicing interval was originally planned for each two years, with a "growth" mission separate from the servicing mission for some platforms.3 "Growth" of the platform would include the addition of new instruments, and any required support structure or engineering, and could also include upgrade (typically by removal & replace ment) of instruments already on the platform. In consideration of several factors including STS schedule constraints at the WTR, it became apparent that a growth mission is more efficiently combined with the servicing mission. The platform and instrument complement (instrument carrier or payload module) are to be designed for survival for up to three years without servicing to allow for contingencies in the servicing schedule.

The altitudes that represent the EOS-NOAA convergence of science synergism interest are not feasible for the STS Orbiter to place or service platforms in-situ. Therefore, alternatives to STS Orbiter in-situ servicing of these platforms have been been explored. The goals in the selection of alternative servicing methods are to minimize service time (when platform is out of normal operations) and minimize risk to the platform (disturbance, contamination, etc.). In addition, servicing scenarios and configurations should reflect realistic current capabilities, or they must identify drivers for new technology develop ment where current capabilities cannot satisfy the platform platform Initial Operating Capability (IOC) or servicing requirements.

[** Note: Instrument complements of the polar platforms are still in negotiation. and final selection of instruments may include EOS, NOAA, commercial and international users; the selected operational altitudes may also change. This paper addresses only the EOS and NOAA requirements, as other users and their requirements were not yet identified at the time this paper was written.]

I PLATFORM AND SERVICING VEHICLE RENDEZVOUS SCENARIOS

Of primary concern in each servicing scenario is the payload lift performance characteristics of the servicing mission launch vehicle (STS) and, if the OMV is the servicing vehicle, the payload lift capabilities of the OMV. Current STS Payload Integration Plan (PIP) estimates put the STS cargo lift capability for a polar orbit at about 8,745 kg.4 STS developmental estimates for upgraded performance (filament-wound booster casings, etc.), have varied with an estimate in 1984 for 13.700 kg, which declined to an estimate in March 1985 of 12,500 kg.5 In order to understand the users' servicing requirements, it was necessary to use a strawman platform configuration to define the envelope of capabilities, requirements and constraints. As a representative platform, we have used a strawman design that includes a carrier module for the instruments, an engineering module (including solar arrays), and a propulsion system (required regardless of the servicing scenario used). We have outlined a representative mass for the platform at Initial Operational Capability (IOC) of 12,500 kg. This mass does not include associated support equipment (ASE) which stays in the STS cargo bay and supports the platform during launch; the ASE mass is likely to make this platform reference configuration mass even less compatible.

Figure I-A shows how the available instrument payload mass (using our strawman platform and assuming delivery by STS to a 278 km altitude) increases and decreases depending on the desired operational altitude to be achieved and the propellant required for the transfer and drag make-up.6 The estimated servicing mass for the first platform servicing mission is about 12,400 kg, including the carrier and engineering module servicing mass, propellant, ASE, and Extra-Vehicular Activity (EVA) support mass; it does not include the Remote Manipulator System (RMS) mass, (about 500 kg) which is assumed in the STS mass. The first platform servicing mission will include the addition of two instruments and their platform support structure, as well as replacement components, etc. When servicing is complete, the platform will have gained about 2,120 kg additional mass. If the platform is serviced at STS altitudes and has integral propulsion, some additonal mass may be required for extra propellant, over the 2,900 kg used for this scenario.

Three rendezvous scenarios are addressed here:

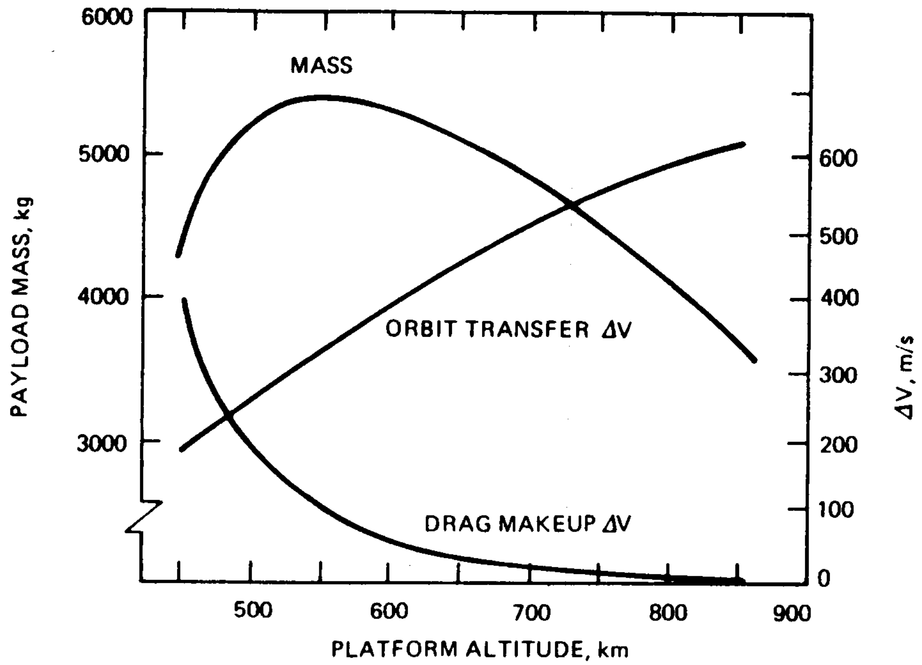
1. The OMV (with "smart" front end) services the platform in-situ at the platform operational altitude.

2. The OMV ("smart" front end not required) retreives the platform and transfers it to STS altitude for servicing.

3. Integral propulsion is designed into the polar platforms for transfer to and from the STS servicing altitude.

1. OMV services the platform in-situ:

This scenario is the preferred servicing method from the point of view of the users. It involves the least disruption of operations, and minimizes the risks to the platform from contamination. However, OMV servicing in-situ requires, as a minimum, a "smart" front end kit with manipulator(s), and assured availability of the OMV at the WTR. User advantages of in-situ servicing relate to reduction of servicing time and risks to the platform. Some types of functions might be done more quickly and easily with the automation and robotics developments on the OMV than they can currently be done with either RMS or EVA (this assumes some of the



- FIXED PLATFORM PROPULSION OPTIMIZED FOR 705 km
- OPTIMIZED STS ALTITUDE
- WORST CASE DRAG FOR 3 YEARS

FIGURE I-A. SERVICING INITIAL LAUNCH CAPABILITY

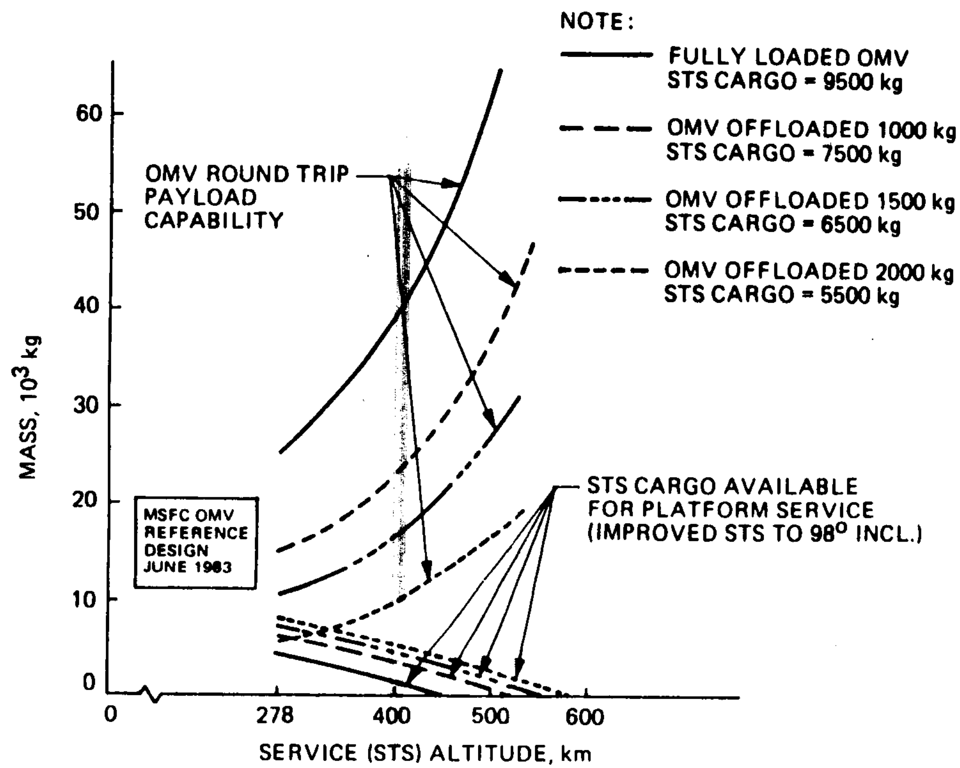


FIGURE I-B. STS/OMV PLATFORM TRANSFER SERVICE TO/FROM STS ALTITUDE

TABLE I-A PLATFORM SERVICE VIA OMV FETCH MODE SERVICE AT SHUTTLE AT 350 km ALTITUDE

	PLATFORM OPERATIONAL ALTITUDE			
		705 km		850 km
• OMV PLATFORM TRANSFER CAPABILITY		12,000 kg		6,900 kg
ENGINEERING MODULE	4200		4200	
PROPULSION	200		200	
AVAILABLE FOR PAYLOAD AND SUPPORT	7600		2500	
• STS CARGO CAPABILITY		10,600 kg		10,600 kg
OMV	4800		4800	
SERVICE MASS AVAILABLE	5800		5800	

optimistic concepts being examined for OMV automation and robotics applications.

With OMV in situ servicing, platform system design may be simplified by eliminating the need for large integral propulsion. Some integral propulsion would be required to provide for altitude maintenance, and a larger propulsion system may still be desireable for back-up STS servicing (detailed trades on this issue would be useful). With a smaller integral propulsion system, there might be some mass margin available for user payloads. Contamination risk during the servicing process would generally be reduced using OMV, compared to the STS environment for both particulate and chemical contaminants. The platform would tend to experience less stress from torque, vibration, and drag than it would with altitude transfer and STS servicing. With in-situ servicing, there is also the possibility of some level of continued operations and data collection during the servicing; a few users have indicated they would prefer that their instruments continue functioning, even if the data were of a degraded quality.

The OMV payload lift performance provides for the capability of taking the greatest servicing mass to the platform operational altitudes. For some optimized conditions of OMV propellant off-loading (Figure I-B), and a slightly higher (307 km) STS injection altitude, the OMV can deliver about 12,000 kg to a platform altitude of 705 km.7 Since the servicing mass is only expected to be between 7,500 kg and 10,000 kg (depending on the mass of OMV support equipment fixtures and propellant), the OMV provides a desireable alternative to the platform transfer with STS servicing.

The potential drawbacks of the OMV servicing scenario involve risks in designing platforms with dependence for servicing on a vehicle that is still only in the conceptual stage. The OMV as a servicing vehicle, with smart front end and manipulator(s), requires development of complex, advanced technologies (if it is to be more than a simple retrieval vehicle). The potential for OMV development delays and incompatibility with stringent polar platform servicing requirements is very great. Current OMV manipulator designs8, specify tip mass load limits of about 700 lbs (318 kg)--one of the growth platform instruments to be attached is 1200 kg (TIMS), and another instrument's replacement component package is estimated at 690 kg (LASA-A laser flash-lamp assembly). So current servicing and growth mission requirements are already not compatible with preliminary OMV design limits. In addition, no assurance can be given for frequency of OMV availability at the Western Test Range (WTR - polar launch site).

Although the OMV will be the preferred servicing vehicle when it has a fully proven smart front end and manipulator(s), it is recommended that the platform be designed for integral propulsion and RMS/EVA standard interfaces as the baseline. It is assumed that the OMV will evolve with systems interfaces compatible with the STS, RMS and EVA standards, the platform will then be assured of servicing in the time frame of the first servicing mission (1995) independent of the development schedule or delays of the OMV, yet compatible with it.

2. OMV retrieval, STS servicing:

The concept of using the OMV as a vehicle to retrieve the platform is viable and can be optimized for certain platform altitudes, configurations and OMV propellant off-loading as shown in Figure I-B. The OMV, its support structure and propellant will be about equivalent to the propellant resupply system required for integral propulsion. Table I-A shows the payload lift performance capabilities for OMV transfer of platforms to/from 705 and 850 km.9

For a platform altitude of 705 km, and with 60% propellant off-load, the OMV could depart from the STS Orbiter at 350 km, retrieve a platform of about 9,000 kg mass, transfer to the Orbiter altitude, then return the augmented and serviced platform, with up to about 3,000 kg additional mass, to its operational altitude and still retain sufficient propellant to return to the Orbiter.[10] In the first platform servicing scenario, the growth platform mass of about 14,620 kg exceeds the lift capability of the OMV for replacement to any orbit above 700 km. In general, the OMV performance begins to fall off for altitudes above 700 km. For transfer between altitudes of 350 km (optimum OMV deployment from STS) and 850 km, the OMV can only carry 6,900 kg of payload mass.

Other potential problems may tend to off-set retrieval advantages of platform propellant reduction and lift capabilities of the OMV at lower altitudes. Risk of both contamination and disturbance is increased through exposure to both STS and OMV environments. In addition to the time required for actual servicing of the platform, there would be time added waiting for OMV transfers, OMV docking and deployment times to-from the platform, as well as the process of STS capture, berthing and deployment of the platform, and re-berthing of the OMV in the STS cargo bay. The servicing mission of the STS may need to be extended, which raises considerations of both mission costs, and the reduction of available payload mass on the STS due to additional crew supplies required for extended missions. If the OMV is berthed in the STS cargo bay with the platform during the servicing operation, it may be more difficult and time-consuming to access and service the platform due to the obstruction caused by the berthed OMV, particularly during RMS-only operations. As in the first scenario, the frequency of availability of the OMV at the WTR is unknown. Finally, another problem is that orbit transfer with servicing at a different altitude involves a drift from the selected orbit plane and and equatorial crossing time (nodal crossing). A key element of the EOS and NOAA observational requirements is the sun-synchronous orbit and selected nodal crossing, posing correction maneuver problems for the OMV and platform combination. This problem is discussed in the final scenario.

3. Platform transfers to STS altitude for servicing:

Finally, we look at the scenario of the platform having an integral propulsion system, transfering to the STS altitudes for servicing and propellant resupply, then returning itself to the desired altitude. This scenario may imply the least risk to the user, in terms of simplification of design choices, and confidence that the technology is available and well-understood. The equatorial or nodal crossing time for the platforms is of particular interest to the users, and transfer from the operational orbit will probably require compensatory maneuvers. The local equatorial crossing time is dependent on both the altitude and inclination of the platform. If the crossing time must be maintained when altitude is changed (as for transfer to STS servicing altitude), a plane (or inclination) change must also be made. The magnitude of the plane change correction is dependent on the difference between the two altitudes, the length of time the platform is not in its operational altitude, and the mass and configuration of the platform. The plane correction may be achieved in several ways:

(platform propulsive corrections)

a) Plane/inclination change prior to servicing (e.g. during descent), to maintain equatorial crossing at servicing altitude and return
b) Plane/inclination change after servicing
c) Smaller corrections are made during both descent and ascent

(non-propulsive correction)

d) Transfer to a higher than operational altitude following servicing, allow drift back to appropriate crossing time and descend to normal operational altitude

(other)

e) Return to a different altitude and nodal crossing (only if the users are prepared to accept and/or take advantage of this change...at the present time, most users indicate their instruments will be designed for specific altitudes and viewing angles and cannot accept such a change)

A few users may prefer the opportunity to change altitudes and/or nodal crossing times, although final selection will depend on science synergism studies. While both OMV retrieval and integral propulsion transfer scenarios involve the problem of orbital corrections, there are some important advantages of integral propulsion over OMV retrieval for the current polar platform concepts. In particular, mass performance advantage for integral propulsion increases with operational altitudes above 700 km . Platform exposure to contaminants can be minimized by covering/closing sensitive instruments and apertures, and by possible purge. STS offers a more stable servicing berth during movement and placement of large masses, and has greater capability for altitude maintenance and dynamic control than the OMV. The option is available in this scenario for more thorough check-out of the platform and components, and better on-board analysis and data relay systems. And the STS with EVA capability is more versatile and better equipped for handling contingencies, especially unusual repair or servicing tasks.

Finally, this scenario provides high confidence for availability of the servicing vehicle and simplicity of using current technologies. Realistic scenarios and timelines can be defined now, and used to assist in the process of platform and instrument configuration studies, and there is a good baseline of experience on which to model servicing configuration dynamics and interfaces. The recommended scenario, then, is to incorporate integral propulsion and plan for platform transfer as the primary servicing method, at this time; however, emphasis should be placed on making input to OMV designs and specifications of the smart front end, to ensure compatibility with platform STS/RMS and EVA interfaces. In this way, servicing is assured by STS or OMV.

II. GENERIC AND SPECIFIC SERVICING REQUIREMENTS AND GUIDELINES

The generic servicing requirements for the polar platforms involve the repair or replacement of life-limited or failed components, the addition of new instruments or upgrading of existing instruments, replenishment of consumables, including propellant, and any other servicing tasks that might be required, such as cleaning optics, recalibration or realignment of instruments, etc. The servicing mission for any servicing scenario consists of six phases:

1. Pre-servicing
2. Launch and orbit transfer (by platform, OMV or both)
3. Proximity operations and berthing
4. Servicing
5. Post-servicing deployment
6. Post-servicing orbit transfer and return to operations

Discussion of these six phases includes identification of generic polar platform requirements, as well as guidelines and some concerns.

1. Pre-servicing requirements and guidelines:

   1) Users will be responsible for providing accurate test models of their instruments for integrated (including STS, RMS, EVA and, if appropriate, OMV) system tests and training.
   2) Users or their experiment representatives should be closely involved with platform and service vehicle operations centers in both pre-servicing and servicing operations.
   3) Scheduling and negotiation of support resources (such as TDRSS links) and specific servicing activities should occur in parallel with or as part of the test and training activities.
   4) Pre-servicing checkout tests should be conducted on the flight ~~platform to provide a~~ servicing checkout status baseline and to assure platform readiness.
   5) Servicing mission launch integration and preparations may require participation of platform operations personnel and experiment representatives at the WTR.

2. Launch and orbit transfer requirements and guidelines:

2.1 OMV transfer to platform orbit (in-situ servicing):

   1) Platform performs servicing readiness test, including retraction of appendages, covering/closing of sensitive instruments, as necessary
   2) Platform conducts full power checkout prior to approach of OMV
   3) Instruments and/or components intended for replacement on the platform may need a validation test post-launch

   [NOTE: The OMV as retrieval vehicle scenario has not been addressed further since the first two platforms will have operating altitudes and masses that exceed OMV planned capabilities for retrieval and redeployment]

2.2 Integral propulsion platform transfer:

For integral propulsion and platform transfer to the STS, two operational options need to be considered:

* the platform descends and waits for the STS to launch, or,
* the STS launches and waits for the platform to descend.

It is recommended that the platform prepare for descent, covering or closing sensitive instruments and apertures, retracting appendages or solar arrays, as necessary, and demonstrate readiness with a descent systems test. However, actual descent should not be initiated until the STS is successfully launched and has completed its on-orbit checkout. This recommendation is based on the following:

   1) STS launch delays are frequent and typically are days or even weeks long
   2) Following descent, the platform will have very limited propellant for return to some "parking" orbit
   3) Drag is greater at STS altitudes
   4) Platform may have solar arrays partially or fully retracted to reduce

drag, and battery power will not sustain the platform very long; deploying solar arrays to provide full power will increase drag unacceptably

5) Platform mission objectives will be severely compromised if the platform is out of normal operations more than a few days

The transfer time is estimated to take about 1 to 1 1/2 hours using a medium thrust transfer method.

3. Proximity operations and berthing requirements and guidelines:

3.1 OMV

1) Platform inhibits attitude control system during approach and docking of OMV, and remains dynamically quiescent throughout servicing
2) OMV will provide a full video scan of the platform prior to docking
3) OMV will provide a minimum of 1.5 Kw power and data relay (rate TBD) while docked with the platform
4) Platform will be equipped with multiple docking fixtures, if required to allow OMV and its manipulator(s) to access all portions of the platform that require servicing
5) Checkout will be conducted following OMV berthing

3.2 STS

Once the platform has achieved the STS altitude, it may maneuver or remain quiescent, as required by the STS crew and ground operations. The berthing fixture is assumed to be at or near the rear of the cargo bay, and the platform attached to the berthing fixture in a vertical or near-vertical tilt position with the interface at the platform aft end, near the propulsion module (see Figure III-C & III-D). This configuration is consistent with the MMS standard berthing fixture and provides maximum access and range of motion of the RMS. The RMS will be used to grapple and berth the platform, and EVA for this part of the servicing would only occur on a contingency basis. Requirements for the platform include:

1) Platform inhibits attitude control system during approach and docking and remains dynamically quiescent throughout servicing
2) Platform checkout is performed after reaching STS altitude, but prior to STS berthing
3) Power and data relay through the STS berthing fixture, following berthing, with a minimum power level of 1.5 Kw
4) Full video scan of the platform is provided during rendezvous and berthing, and for all other servicing activities
5) Platform systems verification is performed following berthing
6) Sun-avoidance and thermal balance will be provided by the servicing vehicle (tolerances TBD)

4. Servicing :

4.1. Generic servicing activities:

1) Carrier, or instrument module servicing
   - add or remove instruments
   - attach additional carrier module support structure or equipment
   - add, remove, replace components (to replace life-limited or failed components, or to enhance capabilities based on growth

or new technologies)
- replenish consumables
- clean, calibrate or realign, as feasible or necessary

2) Engineering module servicing
   - add, remove, replace components (to replace life-limited or failed components, or to enhance capabilities based on growth or new technologies)
   - replenish propellant
   - attach additional structure or equipment, if necessary, to support carrier module growth

4.2 Generic servicing requirements:

The following are servicing requirements derived for a baseline servicing scenario of platform transfer and STS servicing. However, these requirements may also be applicable for other servicing scenarios. These requirements should serve as guidelines and drivers on enhancements/developments to servicing vehicles and on platform development:

1) Platform solar arrays may be "feathered", partially retracted, or fully retracted, as necessary, for descent and servicing; movement of the arrays will be inhibited during servicing, unless required for access.
2) Platform nominal operations will be suspended from pre-servicing readiness checkout until after the return of the platform to operational altitude.
3) Instruments with sensitive optics, apertures, or surfaces will be covered or closed, as necessary, prior to servicing.
4) Instruments will be put on minimum or zero power levels during the servicing mission, except for required checkouts.
5) The servicing vehicle will provide power to the platform during servicing (while attached).
6) The servicing vehicle will provide telemetry and command relay for the platform during servicing (while attached).
7) The servicing vehicle will provide video monitoring of the servicing operations and video inspection of the platform; this data will be made available to platform operations and the users in real time and in a non-real time archive or review product.
8) The total servicing mission duration (time of platform out of service) shall be nominally 3 days, but not longer than 8 days. In every scenario, the platform must not remain below 300 km for longer than 20 days, due to drag makeup constraints.
9) Both real time monitor data and non-real time data products will be supplied by the servicing vehicle operations center to the platform operations center and to platform instrument representatives; this data will include both platform status and configuration parameters, and ancillary data.
10) One single access (SA) TDRSS link is required for platform command and telemetry, in addition to whatever links are required for the servicing vehicle.
11) Platform will be equipped with STS and RMS compatible grappling fixtures and receptacles/connectors for power and telemetry umbilicals or interfaces; the platform will also meet man-safe EVA standards.
12) Development of the OMV and "smart front end" kit shall include interfaces compatible with the platform's STS/RMS interfaces.

13) The platform will remain quiescent during servicing and altitude maintenance and thermal control will be provided by the servicing vehicle while the platform is attached.

14) Vibration, disturbance and plume impingement by the servicing vehicle will be kept within limits to be determined.

15) The platform will be equipped with STS/RMS standard berthing and grapple fixtures and interfaces; and, if OMV servicing is assured, may also be equipped with one or more OMV docking fixtures; EVA hand/foot-holds will also be provided for contingency handling.

16) STS RMS will have small grapple end effector capability, and will be equipped with force-torque sensors with feedback, and optical or other alignment/positioning sensors.

4.3 Detailed servicing activities and requirements - first servicing mission of first platform used as representative details:

The first servicing mission is planned to both service and augment the first platform in 1995. This platform will nominally be at an 824 km orbit. A representative manifest (at this time, only EOS instruments identified) includes:

Advanced Data Collection/Location System - ADCLS
Correlation Radiometer - CR
Earth Radiation Budget Experiment - ERBE
High Resolution Imaging Spectrometer - HIRIS
Light Detection and Ranging (LIDAR) Atmospheric Sounder and Altimeter - LASA-A
Moderate Resolution Imaging Spectrometer, Tilt mode - MODIS-T
Moderate Resolution Imaging Spectrometer, Nadir - MODIS-N
Nadir Climate I/S - NCIS
Special Sensor Micro Imager - SSMI

During the servicing, two additional instruments will be attached, as well as extra platform support structure. This becomes the "growth" platform. The new instruments will be the Advanced Mechanical Scanned Radiometer (AMSR), which will weigh approximately 320 kg, and the Thermal Infra-red Imaging Spectrometer (TIMS), weighing about 1200 kg. The other large servicing mass (aside from platform propellant) is a planned replacement of the laser flash-tube assembly and associated electronics module for the LASA-A instrument, which may have a mass of about 690 kg. The additional structure for the two new instruments is estimated to have a mass of about 600 kg.

The timeline in Figure II-A depicts a sample servicing scenario for this representative mission, with platform integral propulsion orbit transfer and STS servicing. In this sequence of events, RMS is assumed to be the primary servicing tool, with EVA as back-up for most servicing functions and to assist or enhance the servicing process. Times for RMS and EVA activities are based on current experience with some allowance for improvement due to maturation of the STS servicing technology and personnel experience, and for a few enhancements, such as improved RMS control software, force-torque sensors and feedback, and targetting/alignment sensors.

The events required to complete the representative platform servicing scenario and growth mission are as follows (the order of events may need further study to optimize the sequence):

1. Platform performs pre-servicing checkout and readiness tests.

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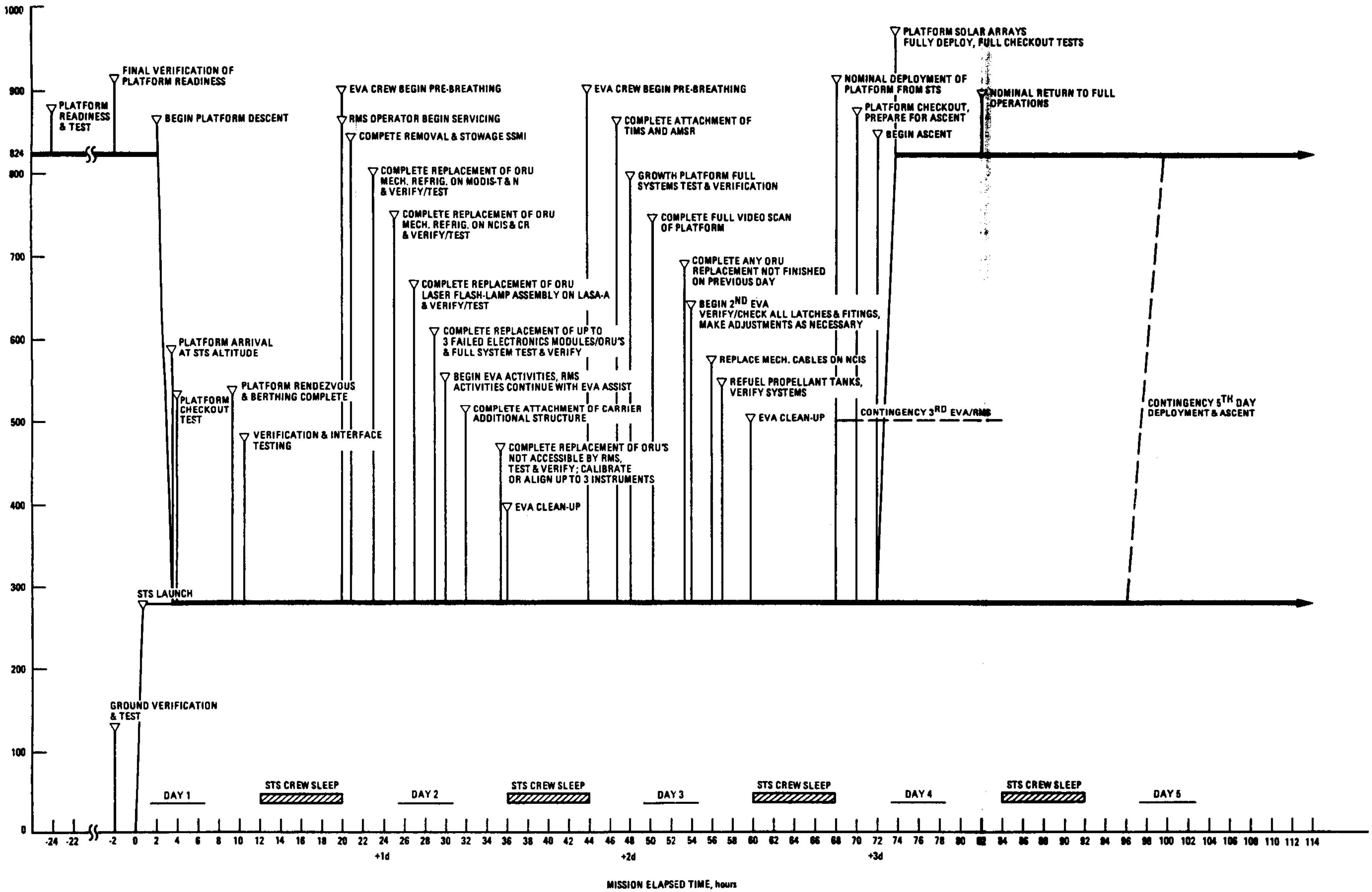

Figure II - A 297

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2. Platform retracts appendages and covers/closes sensitive instruments.
3. STS launch, on-orbit checkout; go for platform descent confirmed.
4. Platform descends/transfers to STS orbit.
5. Platform performs system checkout prior to approach of STS.
6. STS proximity operations, platform grapple and berthing.
7. Platform system checkout in berthing configuration.

(begin servicing)

8. Remove and stow SSMI
9. Replace mechanical refrigerator units on MODIS-T, MODIS-N, HIRIS, and NCIS (unless cryogenics used), and perform verification test.

NOTE: Advanced cooler design technology may eliminate this requirement.

10. Replace laser tube and flash-lamp assembly, and up to two associated electronic modules on LASA-A, and perform verification test.

NOTE: Advanced design technology may eliminate this requirement, or may reduce the mass to be replaced; servicing may consist of upgrade of equipment more than simple replacement.

11. Refuel propellant modules, or replace with loaded tanks, and perform propellant system verification test (line pressure, etc.)
12. Replace failed or life-limited engineering module components (perhaps about 6 units), and perform verification tests.
13. Replenish cryogenic fluids for NCIS (if no mechanical refrigerator), and perform verification test.
14. Attach additional carrier structure.
15. Attach TIMS, and perform system verification test.
16. Attach AMSR, and perform system verification test.
17. Purge platform, if necessary.
18. Perform integrated systems verification tests.
19. Deploy platform.
20. Full platform checkout under platform power.
21. Platform ascent to operational altitude.
22. Platform deployment of solar arrays and operational checkout.
23. Verification of contaminant dissipation (may not be possible).
24. Uncovering or opening of sensitive instruments.
25. Normal operations resumed.

III. PLATFORM CONFIGURATIONS AND SERVICING VEHICLE COMPATIBILITY

Several configurations for the platforms have been examined for the purpose of better understanding the user servicing requirements and the potential for servicing vehicle contraints. The strawman configurations are meant to assist in providing guidelines to the users and contract managers. Reference configurations that satisfy normal operational requirements may be incompatible with certain servicing scenarios and conditions. Figures III-A and III-B depict possible configurations of the first two polar platforms and their instrument complement. These configurations satisfy current identified operational requirements and are compatible with STS servicing. They do not depict the additional docking fixtures that would be required for OMV servicing. Figures III-C and III-D show how these platforms might look during STS servicing.

The platforms were configured to fit within the STS cargo bay at IOC, and include retractable solar arrays. The box-beam concept was chosen because of its lower mass, good heat rejection capabilities, better growth potential and less obstruction to most instrument fields of view. The addition of the TIMS and AMSR in the first servicing mission, puts the upper end of the platform nearly out of

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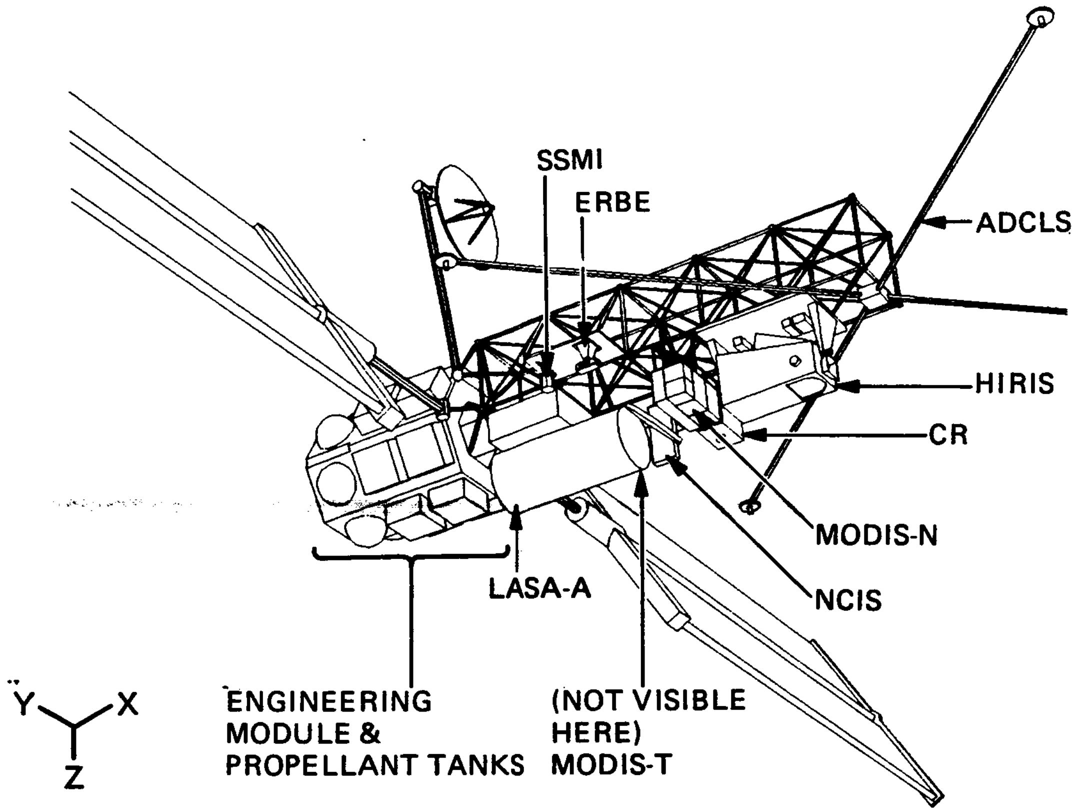

FigureIII-A. Polar Platform 1: IOC (1993) Strawman Configuration

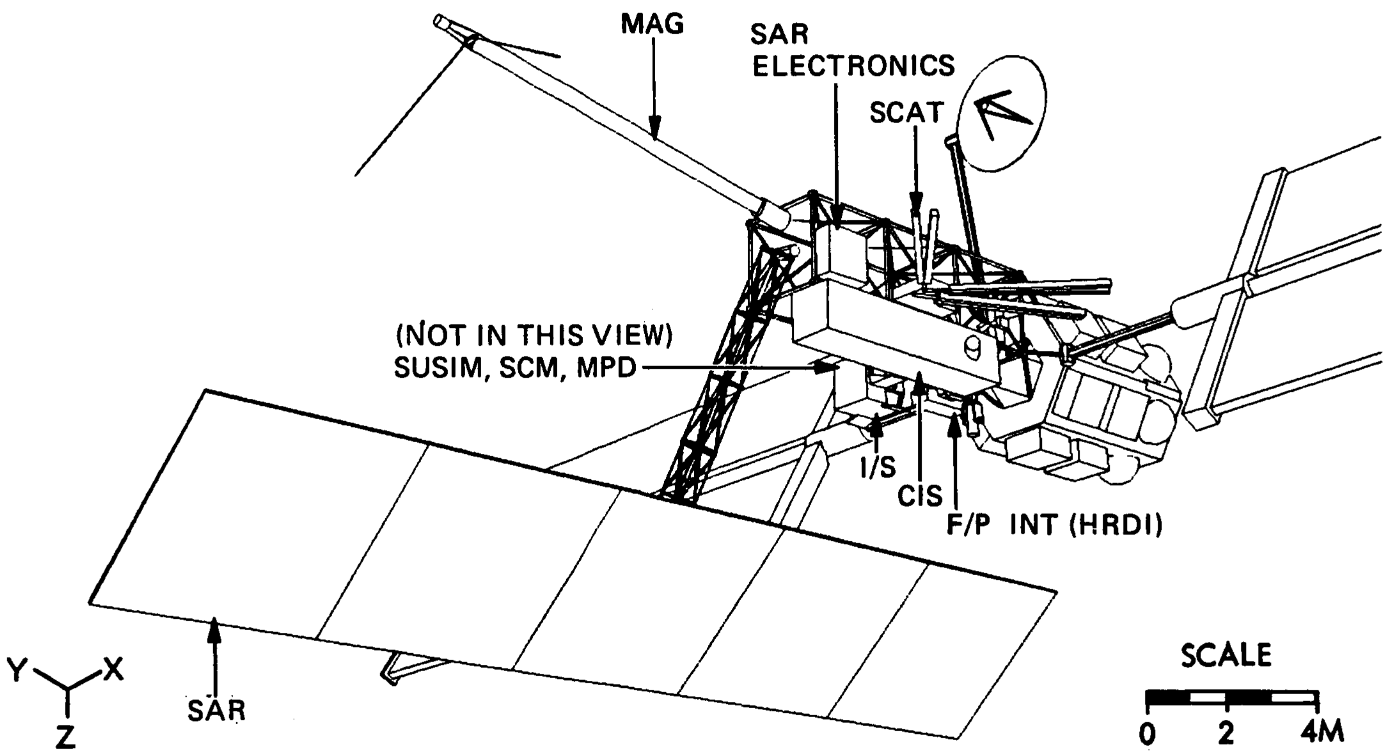

Figure III-B. Polar Platform 2: IOC (1994) Strawman Configuration

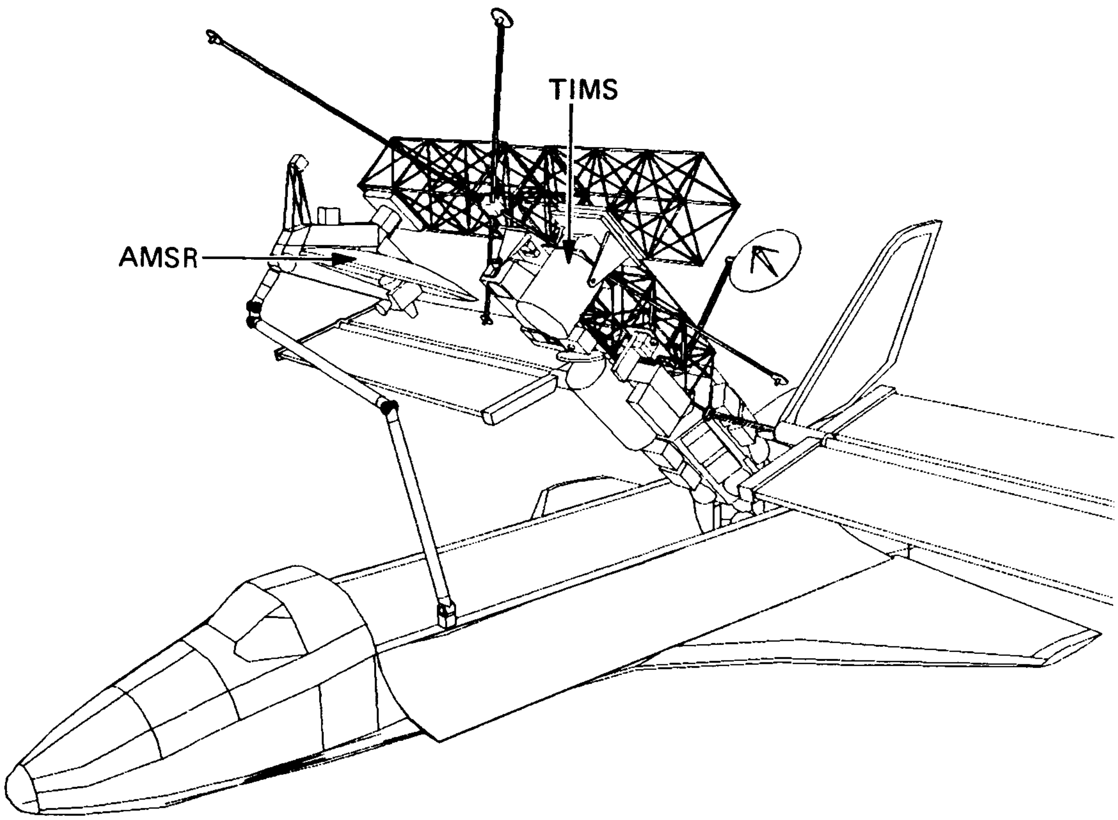

Figure III-C. Polar Platform 1: Growth Strawman Configuration at First Servicing (1995)

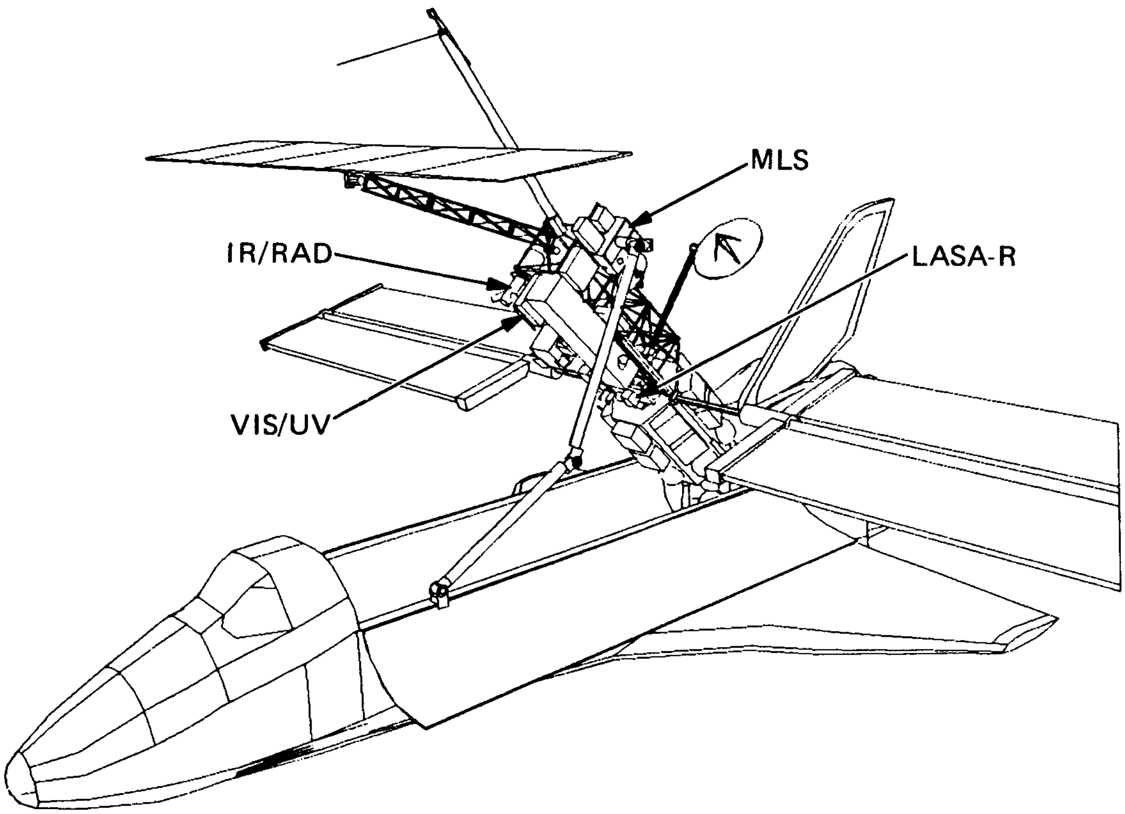

Figure III-D. Polar Platform 2: Growth Strawman Configuration at First Servicing (1996) the radius of RMS access.

A similar problem exists with the configuration possibilities of the Platform 2 with its current tentative EOS manifest. STS or OMV servicing becomes difficult with the complex array of instruments, especially the SAR, even when it is folded/retracted; alternatives include placing the SAR on an articulating boom that can be moved to one side or to the back of the platform. Another alternative is to make the SAR removable for servicing, but this would require more servicing vehicle activity, and is less desireable to the experimenters.

Not shown in these reference configurations, but under consideration, is the addition of small video cameras on the platform. These could assist in both operational inspection and in alignment or positioning during servicing. In order to futher facilitate targetting, the platform could be equipped with its own optical sensors calibrated for a variety of RMS positions, and feed-back to the RMS operator/system. Also, not shown, is a possible manipulator; some OMV configuration concepts have assumed a manipulator on the platform.

Platform power and data system compatibility should pose no problem for STS interfaces, except in the case where SAR data collection during servicing is planned. Standby power requirements for the first platform are estimated between 600 watts and 1.5 kw, which is within the Orbiter reference capability. Platform status parameters will feed through the power and data interface and be relayed to the ground with Orbiter data. If SAR is expected to operate during servicing, even for short periods (10 min.), it will require 4-9 Kw during operational cycles.

RMS interfaces in the form of grapples will be provided on all components or instruments expected to be replaced or repaired. The platform will also be equipped with EVA hand and foot-holds, and tether restraint fixtures.

Compatibility with the OMV for retrieval will require one or more docking fixtures. Compatability with the OMV as a servicing vehicle requires multiple docking fixtures, and may require a manipulator on the platform in addition to OMV manipulators.

IV. ASSUMPTIONS AND ISSUES FOR FURTHER STUDY

IV.1 The following assumptions are made about platform servicing capabilities, that apply to the various servicing scenarios:

1. At least the first two polar platforms will have integral propulsion, and will be serviced by STS.
2. STS servicing altitude will be optimized for the servicing payload, the platform IOC mass and configuration, and transfer requirements.
3. Platforms will have on-board guidance for steering during propulsion maneuvers.
4. STS servicing will be a mature technology by 1990's.
5. Servicing STS will have improved performance, including filament wound SRM cases, lightweight external tank; maximum crew will be 5 persons for a 7 day maximum mission, with RMS, and 2 person EVA supplies for a maximum of 2 EVA's.
6. Instruments and modules requiring servicing or replacement will be designed for RMS and EVA handling.

7. Users will provide test models and servicing guidelines for pre-servicing test and training.
8. The servicing mission will have priority for TDRSS and ground resource scheduling.

IV.2 Some additional issues have been identified that could have significant impact on polar platform design and operational requirements. These issues need further study to fully understand what requirements are needed, or what constraints may be imposed in the platform and servicing vehicle development and implementation.

1. Current STS lift capabilities, as defined by the PIP documentation, do not meet platform requirements for IOC (placement in a single launch) or servicing for WTR launch to polar orbit in a single launch.
2. OMV lift and operational capabilities are not defined, and its availability for WTR and polar operations is unknown.
3. TDRSS and ground support capabilities, as presently defined, are limited: 2 SA links will be required for the Space Station during normal operations, 1 or 2 SA links are required by STS during STS missions, which are planned to occur up to twice per month during the 1990's. Only 1 SA link is planned for all the platforms and Earth-orbiting spacecraft to share.
   a. Over-commitment of TDRSS channels should be investigated, as well as link performance in a variety of conditions.
   b. Data throughput to the users (both real time and non-real time data products) is not well understood, nor are the data system interfaces for "tele-science", and user involvement during servicing operations.
   c. Location and coordination plans for platform, servicing vehicle and instrument operations centers are undefined.
4. A purge of the platform following servicing to clear contaminants may be required; techniques for such an activity need to be defined.
5. Specific data parameters in STS and/or OMV ancillary data required by platform operations and users need to be identified.
6. Techniques need to be studied for accessing various parts of the platforms, particularly the growth versions.
7. Some platform positions during servicing may block STS to TDRSS data relay, due to the size and configuration of the platform and the possible need for sun-angle thermal control maneuvering during servicing. The impact of this on the servicing activity sequence needs to be examined, as well the possible advantage of having the STS crew perform some of the platform command sequences, such as checkouts after component replacement or servicing.
8. Charging effects during servicing should be further studied for the variety of platform and servicing vehicle configurations.
9. EVA radiation hazard in the polar orbit environment is not well understood and may constrain EVA servicing scenarios.
10. STS video monitoring of the servicing activities may be blocked by large instruments being moved by the RMS. The possiblity of platform video cameras with a feed to the STS monitors and RMS operator should be investigated as a means to enhance servicing.
11. The ability to control the docked servicing vehicle and platform pair must be studied for a variety

of configurations and conditions.

The research described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

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Satellite Servicing System
Ground Operations

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CS-SED-32
Kennedy Space Center, Florida

18 September 1985

Ground processing of a Satellite Servicing System (SSS) is obviously an item of interest to the system designer. It should also concern the system user - the customer whose satellite is being serviced - since it may result in an inability to provide the on-orbit service in a timely manner. As an example, if the ground processing time for an SSS takes 6 months, and the user satellite only has a three month lifetime after certain repairable failures, the Satellite Servicing System won't do much good. If the ground processing of an SSS requires personnel who won't be available in a timely manner due to the lack of travel funds or other similar problem, again the system is useless. If the servicing system ends up being a point design that costs almost as much as the satellite being serviced, its utility is questionable. If no ground facilities exist to process the servicing system, it won't be able to do its job on orbit. It therefore is worthwhile to look at those aspects of ground processing of an SSS - the processing flow or schedule, processing team philosophy, facility requirements and availability, and shared servicing systems.

The processing schedule for an SSS obviously is affected by many factors. Undoubtedly, the initial flow of any system will be longer than that for subsequent missions. However, the operational flow is probably the most significant from a system design standpoint, since it defines the number of missions that can be flown by one system in one year. As a result, it can be used along with a projection of the annual servicing requirements to determine how many servicing systems are needed. The Mission Overview and Pad Operations schedules, Figures 1 and 2, should be considered a best case estimate. They are our equivalent of the infamous "160 hour Orbiter Processing Schedule", although ours are hopefully more accurate. As you can see, they show a minimum turnaround time of about three months from launch to launch. This should be considered as an absolute minimum turnaround for any one system. A significant portion of this time is related to hazardous Orbiter operations that are required at the Pad. These operations result in a general Pad clear, which precludes concurrent payload ops. Some of the schedule time also includes operations on other elements sharing a given launch with the SSS. It is possible to eliminate most of this time by flying a dedicated mission, but there are obviously significant cost considerations involved in that decision.

Please note that we strongly recommend against doing any unnecessary payload servicing at the Pad. In some instances, such as cryogenic propellant loading, it may be unavoidable. However, in most cases other options exist if the payload is designed properly. With the current cost of serial time operations at the Pad running around $31,000 per hour, and in light of the fact that it takes significantly longer to perform any operation in the Orbiter environment due to the added concern for the safety of the Orbiter and the other payloads, it is easy to see that system design to avoid Pad operations can result in significant savings over the lifetime of an operational Satellite Servicing System.

Note that these schedules assume optimum system design, no system failures, and minimum retesting between launches. If the system design is such that it requires major hardware deintegration between launches, if the hardware used is of low reliability and requires major rework between each mission, or if the checkout requirements include a retest of all components before each flight, the turnaround time could easily reach six months for either vertical or horizontal integration. We are also assuming that recurring CITE testing will not be required. We expect that some form of checkout equipment will be required for the hardware integration phase of system processing. If this is designed properly, it should be able to do an adequate checkout of the payload to Orbiter interface without going into CITE. We do recognize that CITE may be needed for the first mission, if the payload to Orbiter interface is significantly different from what has flown before. We expect to evaluate that situation on a case by case basis.

It is worthwhile to consider the processing team philosophy. For a NASA-owned system, we at KSC expect to pick up the Operations and Maintenance responsibility, with the original design center retaining the responsibility for Sustaining Engineering for the life of the system. We expect that a NASA System would be integrated and serviced by KSC personnel, much as is currently done for Spacelab payloads. Moreover, since the system would not be directly related to the Orbiter Processing flow, we in Cargo Operations plan to do the work. Our people are familiar with the facilities and equipment available at KSC, and, being permanently stationed there, can save NASA significant expenses associated with travel to the launch site. We have successfully processed Spacelabs 1, 2, and 3, and are currently working on SLS-1, EOM 1/2, ASTRO-1, and Spacelab D-1. We have integrated the OSTA-2, OAST-1, OSTA-3, LFC, ORS, and HS-376 SRM partial payloads, and are presently doing the integration and checkout for EASE/ACCESS and MSL-2. We cooperate routinely with all the other organizations at KSC, and can marshal the support necessary to get the job done in a timely manner. Possibly most important, we are familiar with the constraints of working to a launch schedule. As a result, we have a better chance of getting the payload ready in time for a launch than would a team of people who were used to an R and D environment. We do recognize the need for some participation by design center representatives for at least the first mission. At very least, we would expect them to provide inputs to our processing procedures, as well as perform a review prior to procedure release. We also are willing to provide the opportunity for limited offline post-shipment hardware checkout, if needed.

On the other hand, if the Servicing System is owned by private industry, we at KSC would expect them to take the responsibility for its integration and servicing, much as is done for commercial deployable satellites at this time. KSC personnel would still be responsible for the integration of the Servicing System with the rest of the payload complement, as well as the installation of the payload into the Orbiter. As is presently the case, the shift in responsibility would occur during the transfer from the Payload Processing Facility to the Vertical Processing Facility (VPF) or the Orbiter Processing Facility (OPF), as applicable.

A major consideration for any Servicing System has to be facilities at KSC for processing before each flight, as well as for storage between missions. Basic information on existing facilities is available from the Launch Site Accommodations Handbook for STS Payloads, K-STSM-14.1, as well as the Facilities handbooks referenced therein. You can obtain additional information by contacting the KSC Advanced Projects Office, PT-FPO, or the Spacelab and Experiments Division, CS-SED. The attached Facility Utilization schedules, Figures 3 and 4, show the activity that is currently planned for our present Payload Processing Facilities (PPF's). Note carefully that these existing facilities are already almost fully utilized well into the future. These schedules do change frequently as a result of the fluidity of the STS manifest. However, it does not appear likely that future manifest changes will create significant long term openings.

It is worthwhile to break the facility situation into those which are acceptable for processing of hazardous cargo elements, and those facilities which are usable only by nonhazardous (from a ground processing point-of-view) payloads. For a NASA developed SSS, the NASA-owned hazardous processing facilities available include the ESA 60, the Delta Spin Test Facility, the Spacecraft Assembly and Encapsulation Facility (SAEF II), and the Cargo Hazardous Servicing Facility (Figures 5, 6, 7, and 8). Note that none of these facilities are presently used for long term storage of hardware. In fact, most of them are solely used for servicing with propellants, and are not even utilized for hardware checkout. Obviously, this would not be satisfactory for an operational tanker system.

For a privately developed system, the NASA hazardous facilities just referenced are available, and in addition there is the Astrotech facility (Figure 9) located just west of KSC adjacent to the Space Center Executive airport (formerly known as Ti-Co).

For NASA owned nonhazardous payloads, the PPF's available include the O&C building, as well as Hangars AO, AE, AM, and S (Figures 10, 11, 12, 13, and 14). All of these facilities are designed for the complete nonhazardous payload processing flow, including hardware integration and checkout. Note that the O&C building is primarily dedicated to the processing of Spacelab hardware, although we do handle some non-Spacelab equipment.

Privately developed systems would have access to the NASA facilities, with the possible exception of the O&C building, as well as the privately owned Astrotech facility previously mentioned.

Of course, there is the option of building new facilities. Unfortunately, the turnaround time from KSC's decision to request construction funding until the facility is ready for use is currently running at least 5 years. Since such a facility would be a major item in the KSC budget, there always is the possibility that funding would not be approved. Therefore, it is imperative that any NASA organization which plans to build an SSS start discussions with us at KSC as soon as possible to insure that facilities will be available to process your hardware. We are currently doing the preliminary planning for new Space Station facilities. While it may not be practical to combine the Servicing System processing with that of the Space Station, it is at least worth consideration. However, we cannot do that unless we know what your servicing system requirements will be.

For privately developed systems, the option of NASA-built new facilities exists, as well as the potential construction of new private processing facilities. These new buildings could be located off KSC, as is Astrotech, or we are willing to consider a proposal for a long term lease of KSC property for the construction of private facilities.

We would like to make some recommendations to those of you who are interested in Satellite Servicing Systems. First of all, while we appreciate the need for competition and enjoy the earthbound benefits of having a choice of Exxon, Texaco, and so on, we would like to strongly recommend as much commonality as possible. If you go out and build a separate Servicing System for each satellite that wants to be resupplied, you will run into problems with getting processing and storage space at KSC. Additionally, the cost of taking this approach is obviously much more expensive than that of designing a smaller number of systems that can each service a variety of payloads. It therefore behooves you to cooperate as much as possible in the system design. If you are a servicing system user, contact system designers to let them know your requirements. This can help save you part of the cost of designing a system yourself from scratch. If you are designing a servicing system, please contact all other potential users of the service to see if you can accommodate their needs. This has the potential to help offset some of your development costs. Obviously, this also can help us at KSC by reducing the facility requirements. With the Federal budget situation what it is these days, we need to do everything possible to make the most of each NASA dollar.

Secondly, we would like to recommend that those of you who are interested in building a nonhazardous SSS consider using an SPS or MDM pallet or an MPESS for the basic carrier. These items have already been developed, have been successfully used for the HS376 Satellite Retrieval Mission, and are being used for the Hubble Space Telescope Maintenance and Refurbishment Missions. While the HST M&R system is a point design specifically for that use, a similar system should be able to support a variety of on-orbit component replacement requirements. Proper design would allow the system to be reconfigured from on payload to another within the time limits of the previously mentioned processing schedules.

Finally, we want to suggest the benefits of close coordination with those groups who are responsible for the interfacing of your carrier with the STS. If you contact us at JSC and KSC early in the development of your hardware, we can help you come up with a design that will be the most cost effective response to your requirements. On the other hand, if you ignore us until your design is firmed up, you may find that it does not meet the requirements of the STS, or that it simply won't work within the capabilities of our system. That unfortunately will result in significantly increased expenses, and make it more difficult for you to meet your on-orbit needs.

CS-SED
STS-XX/ OV-XXX

SATELLITE SERVICE SYSTEM REFLIGHT MISSION OVERVIEW

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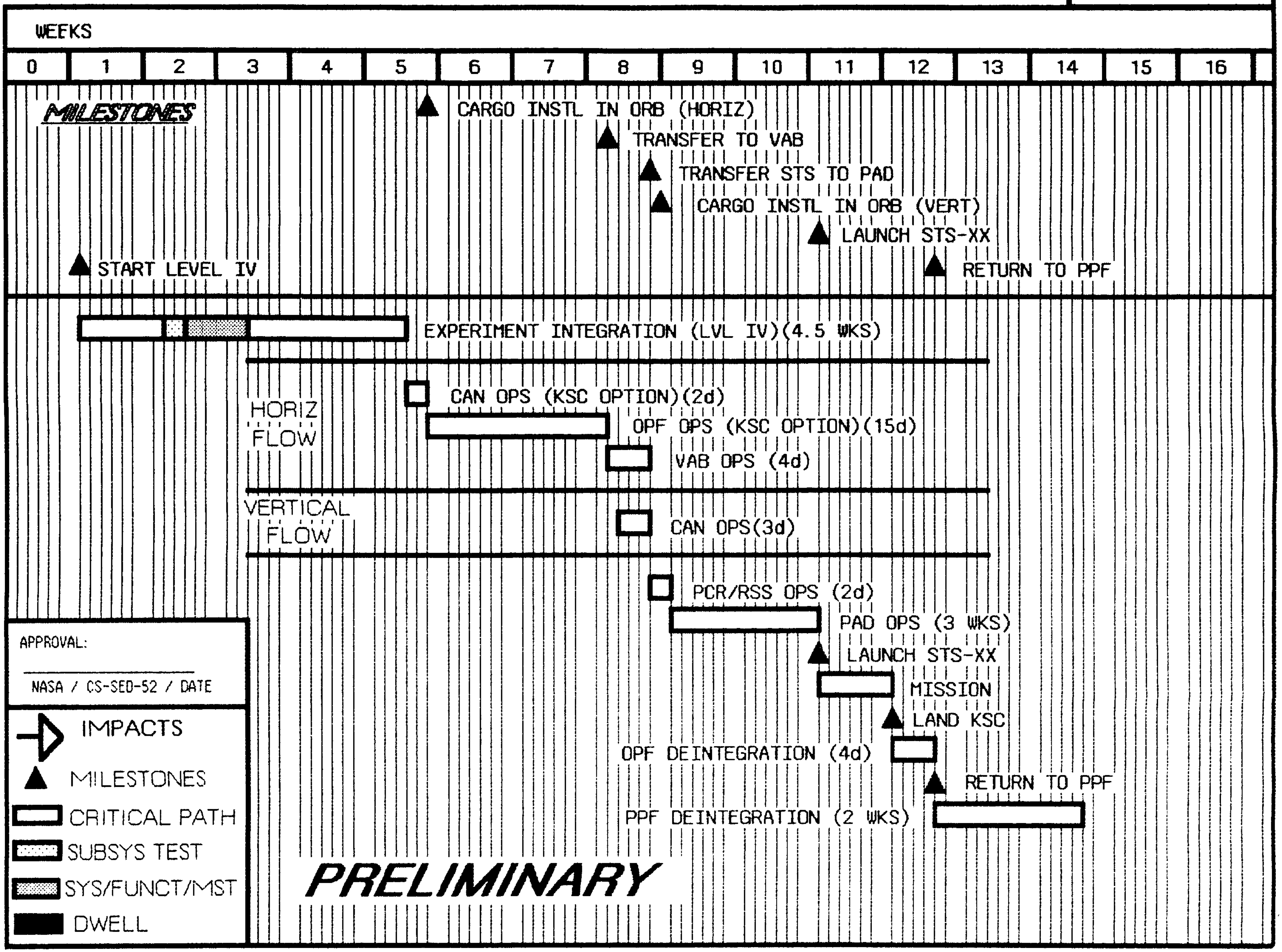
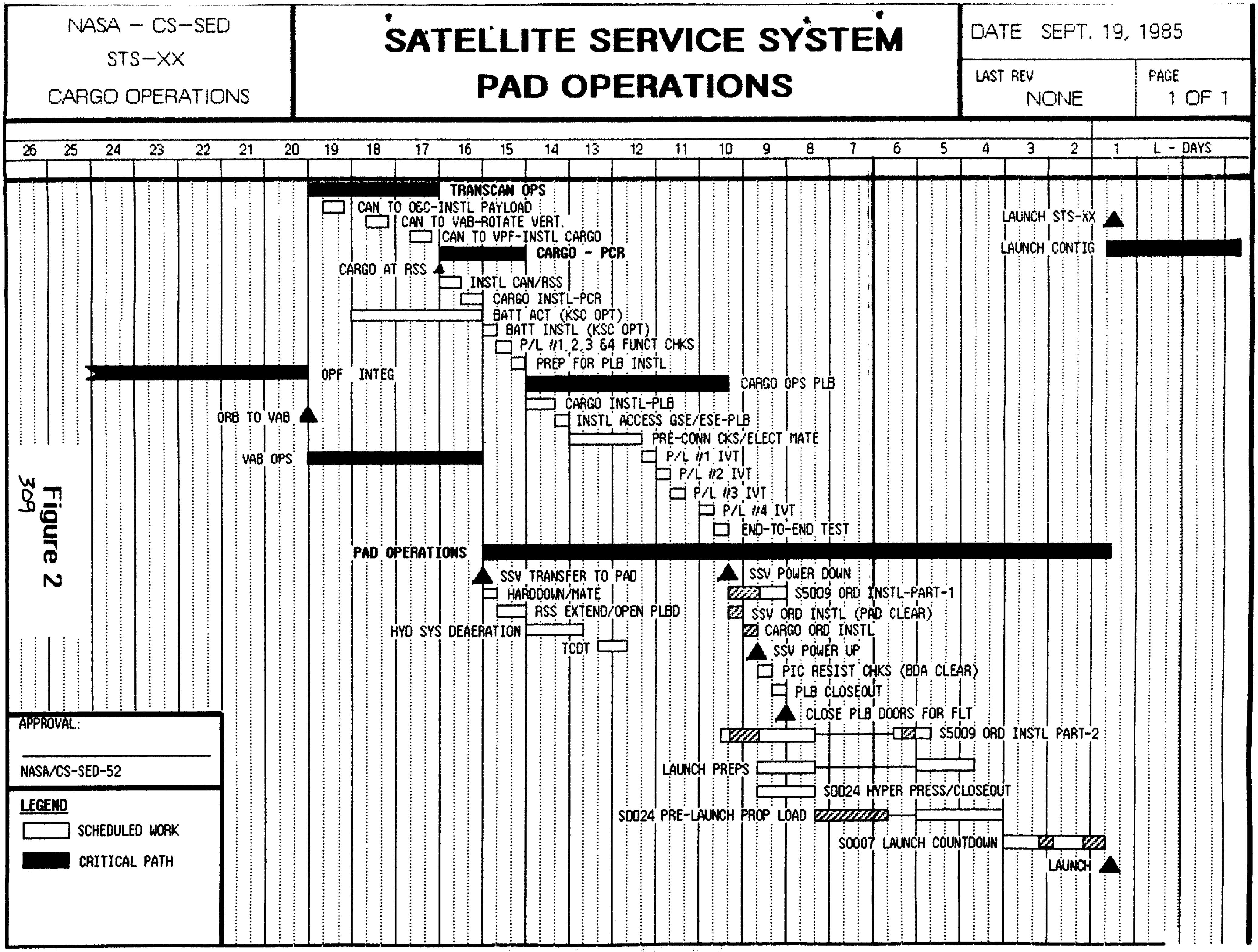


Figure 1


NASA — CS—SED
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CARGO OPERATIONS
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PAD OPERATIONS
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HYD SYS DEAERATION
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CARGO ORD INSTL
SSV POWER UP
PIC RESIST CHKS (BDA CLEAR)
PLB CLOSEOUT
CLOSE PLB DOORS FOR FLT
S5009 ORD INSTL PART-2
LAUNCH PREPS
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LAUNCH
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LEGEND
SCHEDULED WORK
CRITICAL PATH

Figure 2

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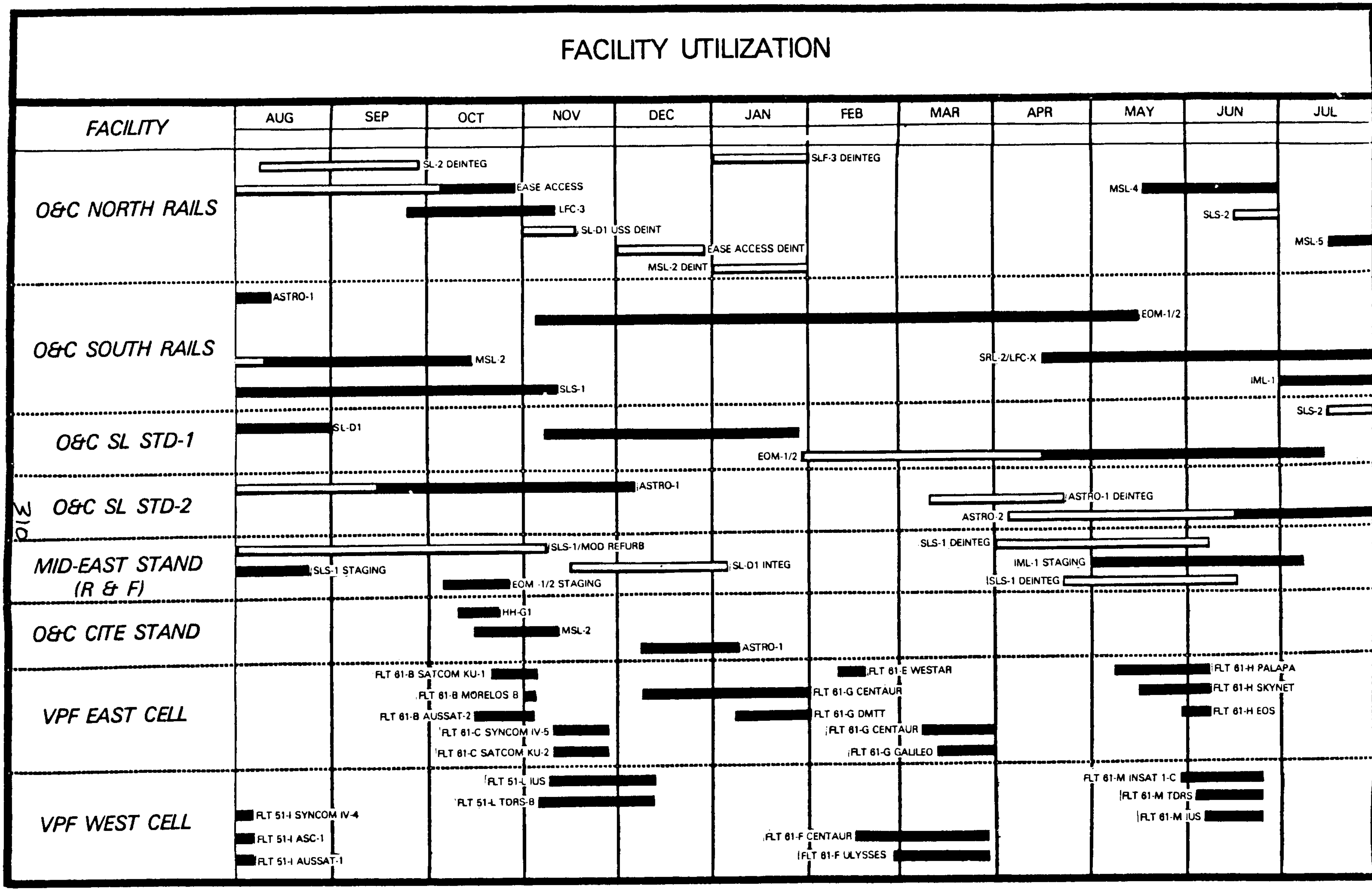

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ASTRO-2
MID-EAST STAND (R & F)
SLS-1/MOD REFURB
SLS-1 STAGING
SL-D1 INTEG
EOM -1/2 STAGING
SLS-1 DEINTEG
IML-1 STAGING
SLS-1 DEINTEG
O&C CITE STAND
HH-G1
MSL-2
ASTRO-1
VPF EAST CELL
FLT 61-B SATCOM KU-1
FLT 61-B MORELOS B
FLT 61-B AUSSAT-2
FLT 61-C SYNCOM IV-5
FLT 61-C SATCOM KU-2
FLT 61-E WESTAR
FLT 61-G CENTAUR
FLT 61-G DMTT
FLT 61-G CENTAUR
FLT 61-G GALILEO
FLT 61-H PALAPA
FLT 61-H SKYNET
FLT 61-H EOS
VPF WEST CELL
FLT 51-I SYNCOM IV-4
FLT 51-I ASC-1
FLT 51-I AUSSAT-1
FLT 51-L IUS
FLT 51-L TDRS-B
FLT 61-F CENTAUR
FLT 61-F ULYSSES
FLT 61-M INSAT 1-C
FLT 61-M TDRS
FLT 61-M IUS

Figure 3

310

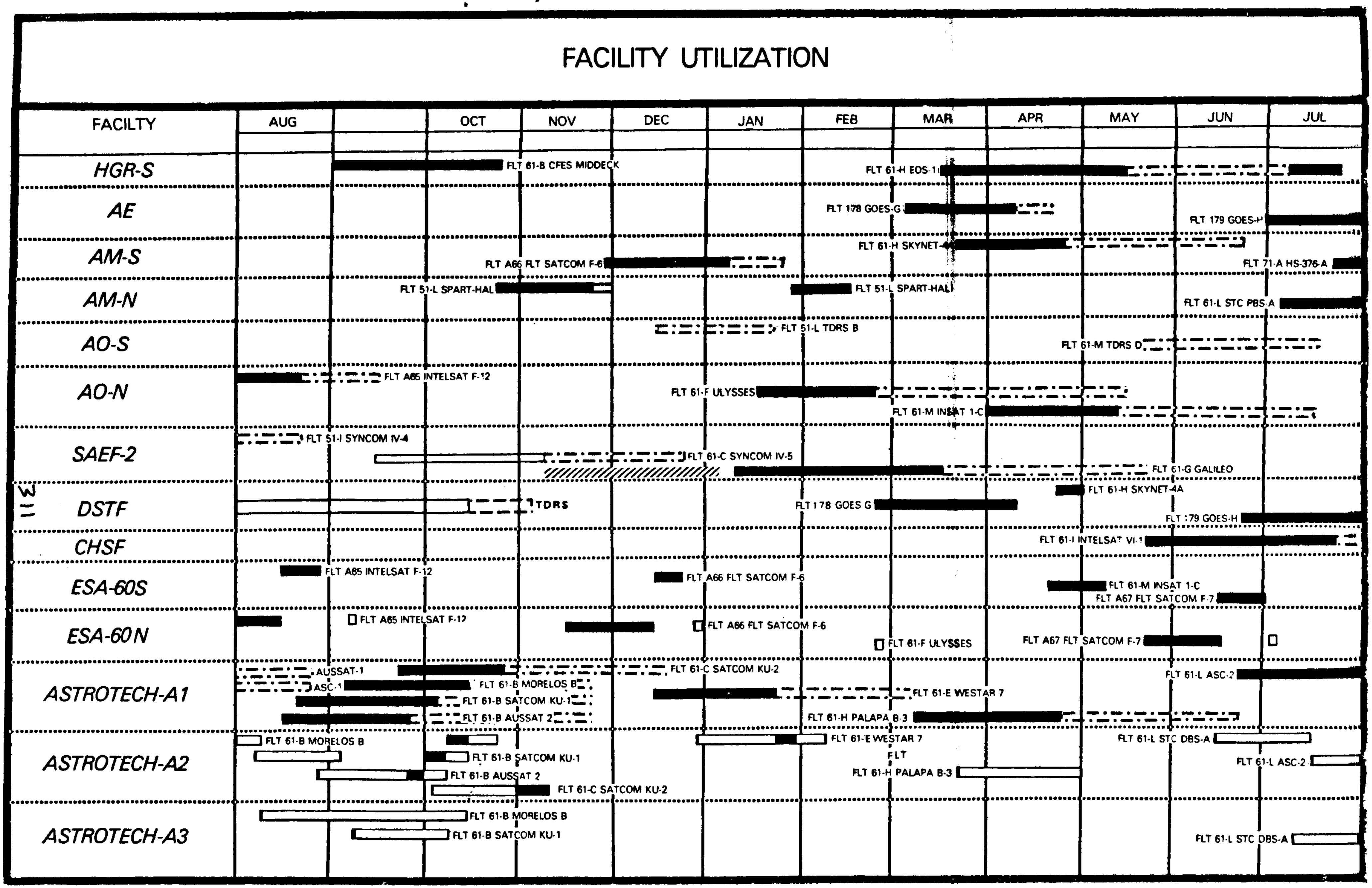
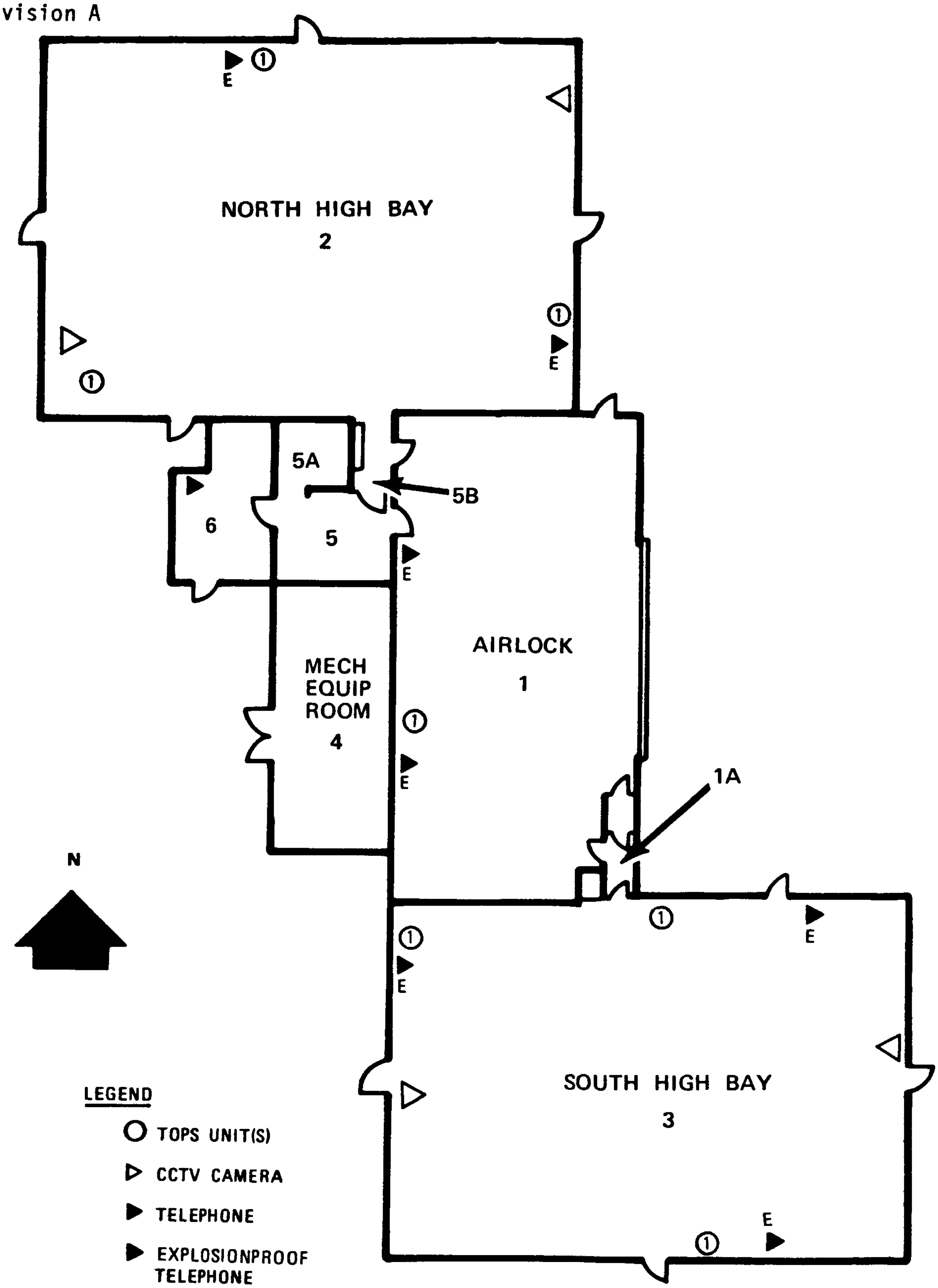

FACILITY UTILIZATION
FACILTY
AUG
OCT
NOV
DEC
JAN
FEB
MAR
APR
MAY
JUN
JUL
HGR-S
AE
AM-S
AM-N
AO-S
AO-N
SAEF-2
DSTF
CHSF
ESA-60S
ESA-60N
ASTROTECH-A1
ASTROTECH-A2
ASTROTECH-A3
FLT 61-B CFES MIDDECK
FLT 61-H EOS-1
FLT 178 GOES-G
FLT 179 GOES-H
FLT 61-H SKYNET-4
FLT A66 FLT SATCOM F-6
FLT 71-A HS-376-A
FLT 51-L SPART-HAL
FLT 51-L SPART-HAL
FLT 61-L STC PBS-A
FLT 51-L TDRS B
FLT 61-M TDRS D
FLT A65 INTELSAT F-12
FLT 61-F ULYSSES
FLT 61-M INSAT 1-C
FLT 51-I SYNCOM IV-4
FLT 61-C SYNCOM IV-5
FLT 61-G GALILEO
FLT 61-H SKYNET 4A
TDRS
FLT 178 GOES G
FLT 179 GOES-H
FLT 61-I INTELSAT VI-1
FLT A65 INTELSAT F-12
FLT A66 FLT SATCOM F-6
FLT 61-M INSAT 1-C
FLT A67 FLT SATCOM F-7
FLT A65 INTELSAT F-12
FLT A66 FLT SATCOM F-6
FLT 61-F ULYSSES
FLT A67 FLT SATCOM F-7
AUSSAT-1
FLT 61-C SATCOM KU-2
FLT 61-L ASC-2
ASC-1
FLT 61-B MORELOS B
FLT 61-B SATCOM KU-1
FLT 61-E WESTAR 7
FLT 61-B AUSSAT 2
FLT 61-H PALAPA B-3
FLT 61-B MORELOS B
FLT 61-E WESTAR 7
FLT 61-L STC DBS-A
FLT 61-B SATCOM KU-1
FLT
FLT 61-L ASC-2
FLT 61-B AUSSAT 2
FLT 61-H PALAPA B-3
FLT 61-C SATCOM KU-2
FLT 61-B MORELOS B
FLT 61-B SATCOM KU-1
FLT 61-L STC DBS-A
W
=

Figure 4



SAB, ESA-60A—COMMUNICATIONS SYSTEMS
FIGURE 7–1

Figure 5

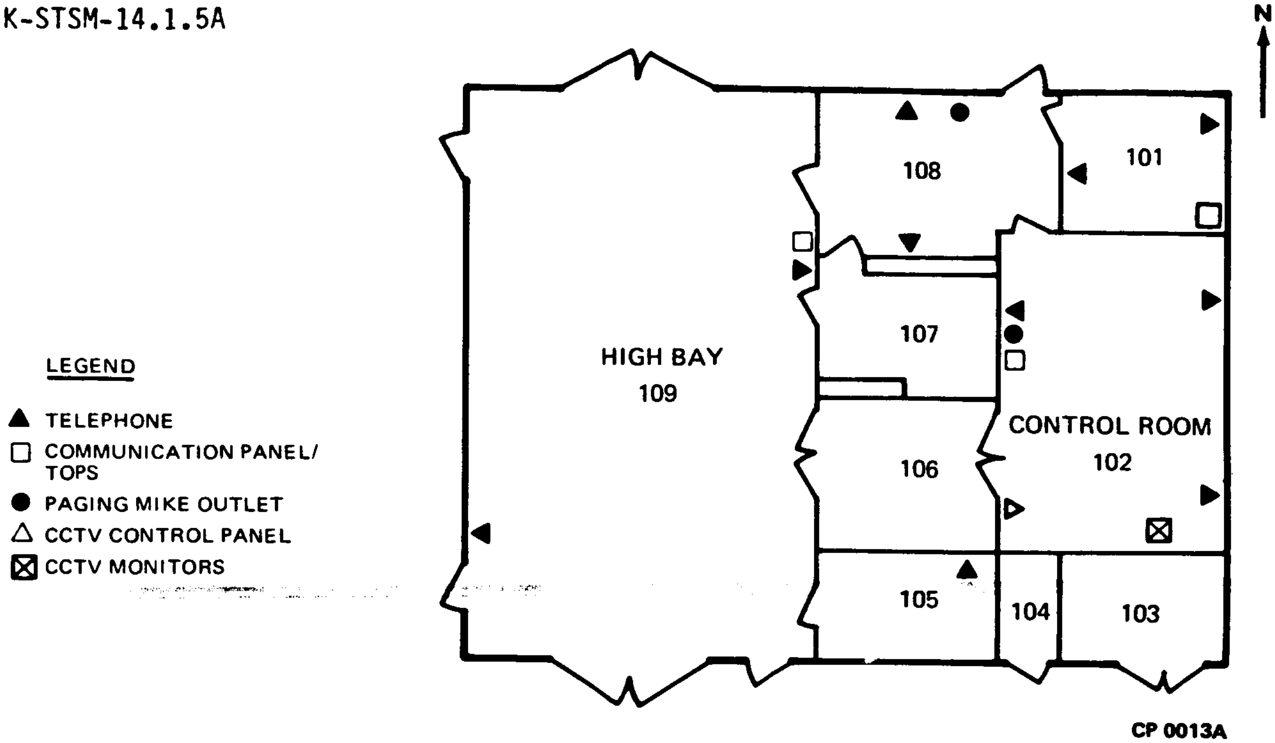
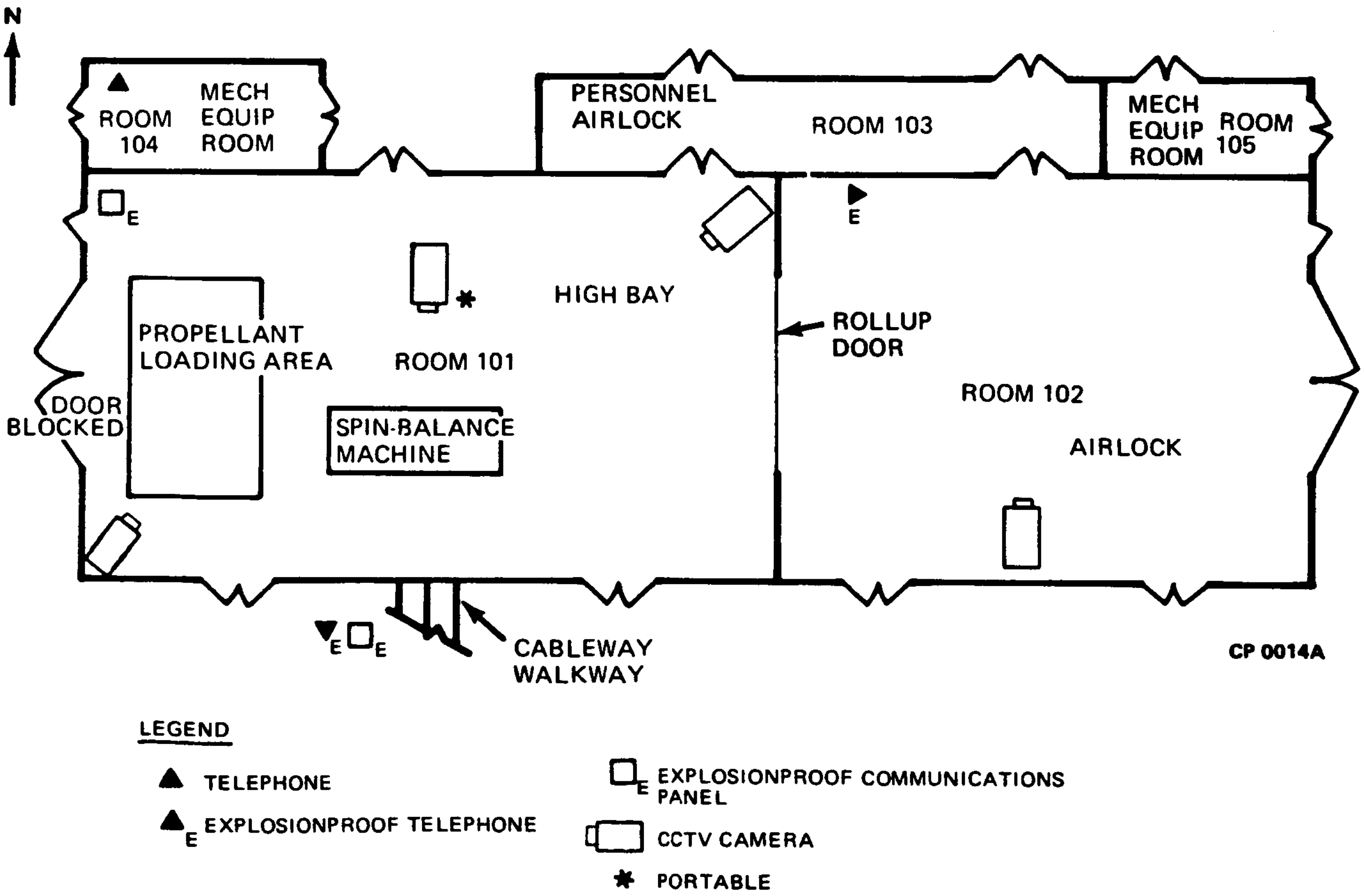


Figure 5-1. Communications System - Control Building, DSTF



COMMUNICATIONS SYSTEM–SPIN TEST BUILDING, DSTF
FIGURE 5–2

Figure 6

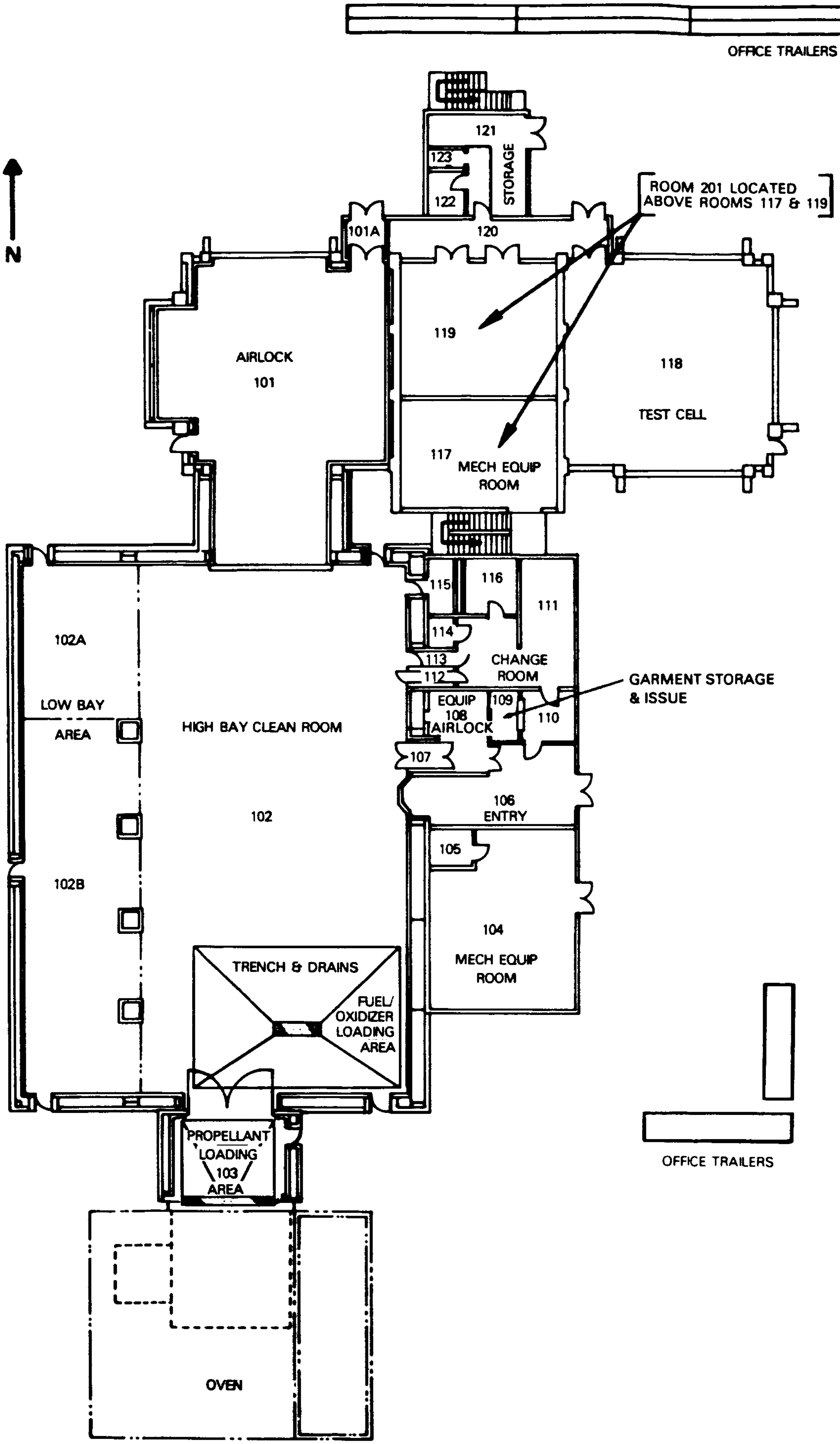
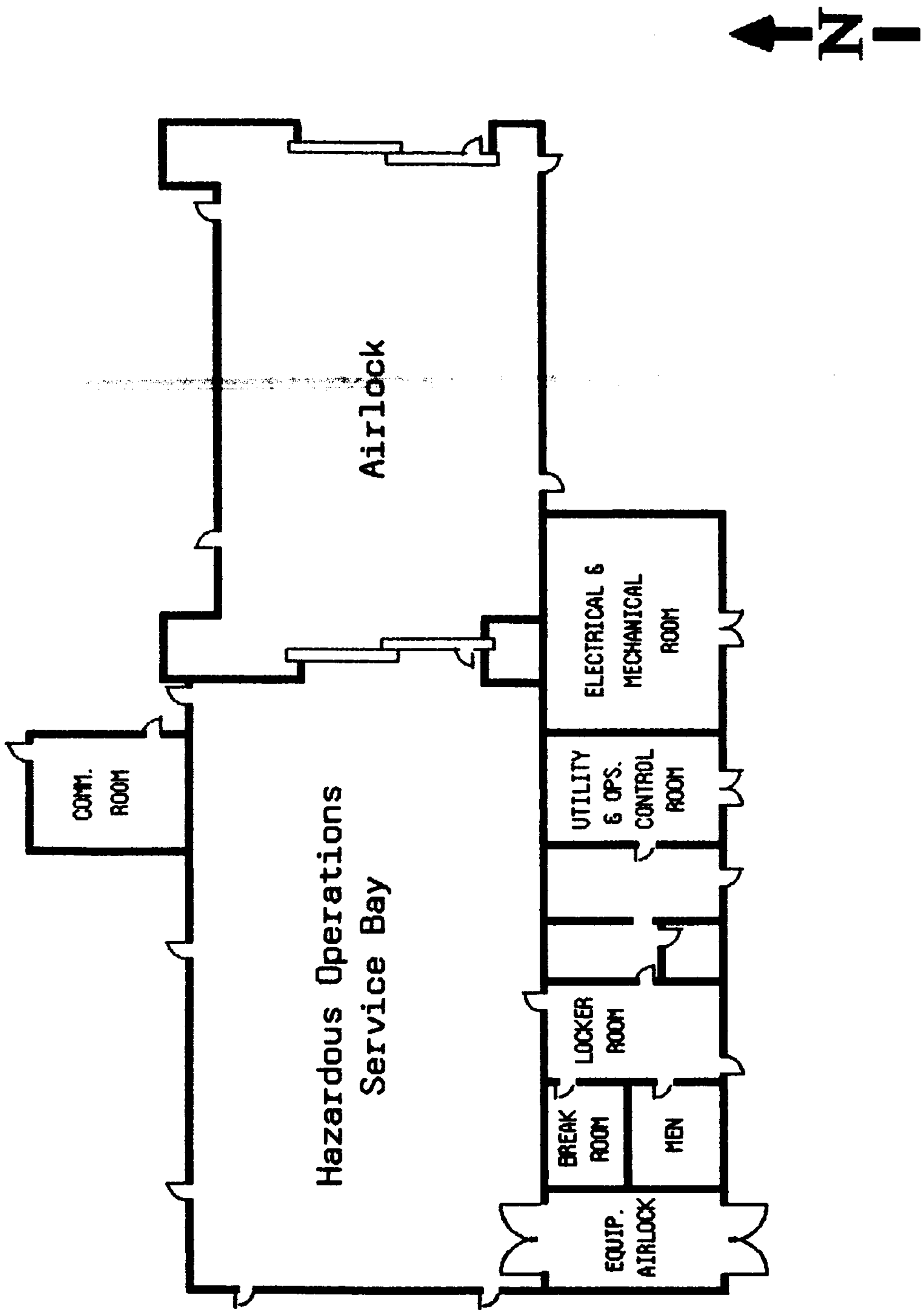


FIGURE 2-3. SAEF-2 Floor Plan

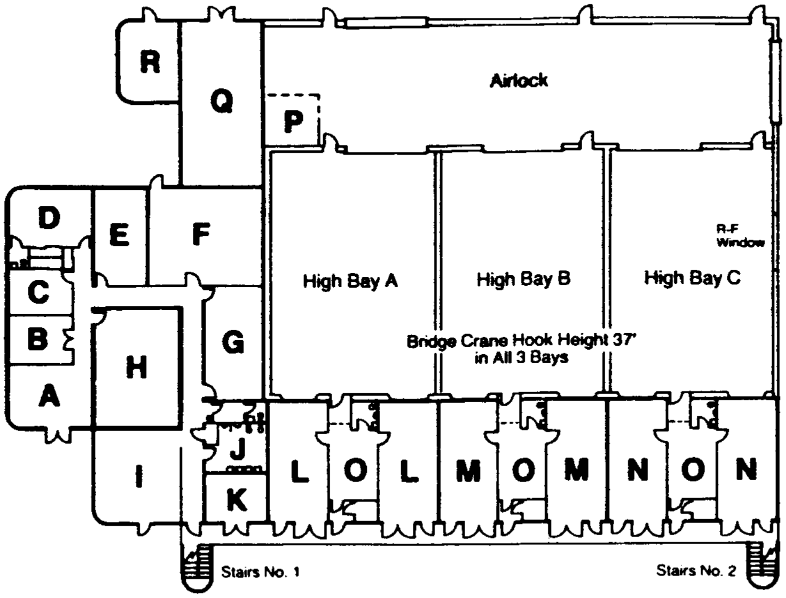
Figure 7



Cargo Hazardous Servicing Facility
Floor Plan

Figure 8

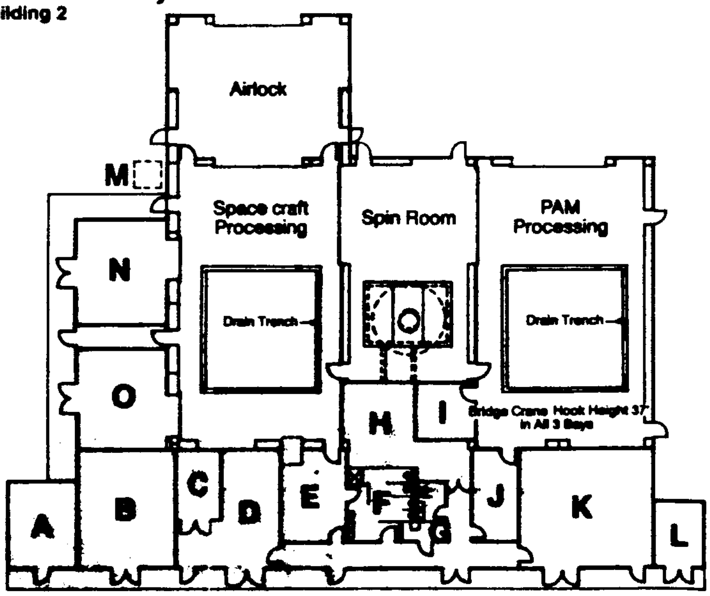
Payload Processing
Building 1
First Floor Plan



Legend

- A Reception Area
- B Reproduction
- C Office No. 2
- D Office No. 1
- E Office No. 3
- F Office No. 4
- G Conference Room
- H Atrium
- I Snack Room
- J Rest Rooms
- K Machine Shop
- L Control Rooms A1 & A2
- M Control Rooms B1 & B2
- N Control Rooms C1 & C2
- O Change Rooms
- P Comm. Room (Above)
- Q. Mech. Room
- R. Electrical Vault

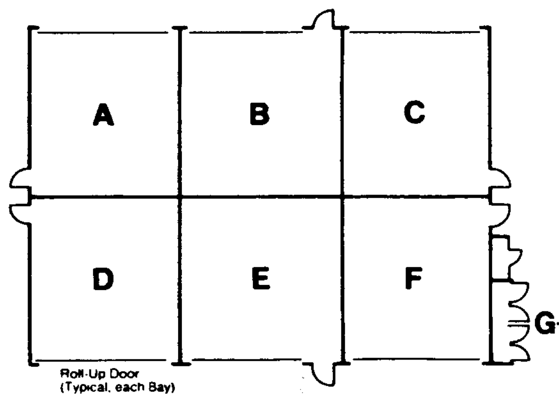
Final Assembly
Building 2



Legend

- A Electrical Vault
- B Mech. Room No. 1
- C Spin Control Room
- D Spacecraft Operations
- E Spacecraft Garment Room
- F Rest Rooms
- G Janitor Room
- H Mech. Room No. 2
- I Plans Room
- J PAM Garment Room
- K PAM Control Room
- L Motor Gen. Room
- M Safety Shower
- N Fuel Cart Storage
- O Oxidizer Cart Storage

Payload Storage
Building 3

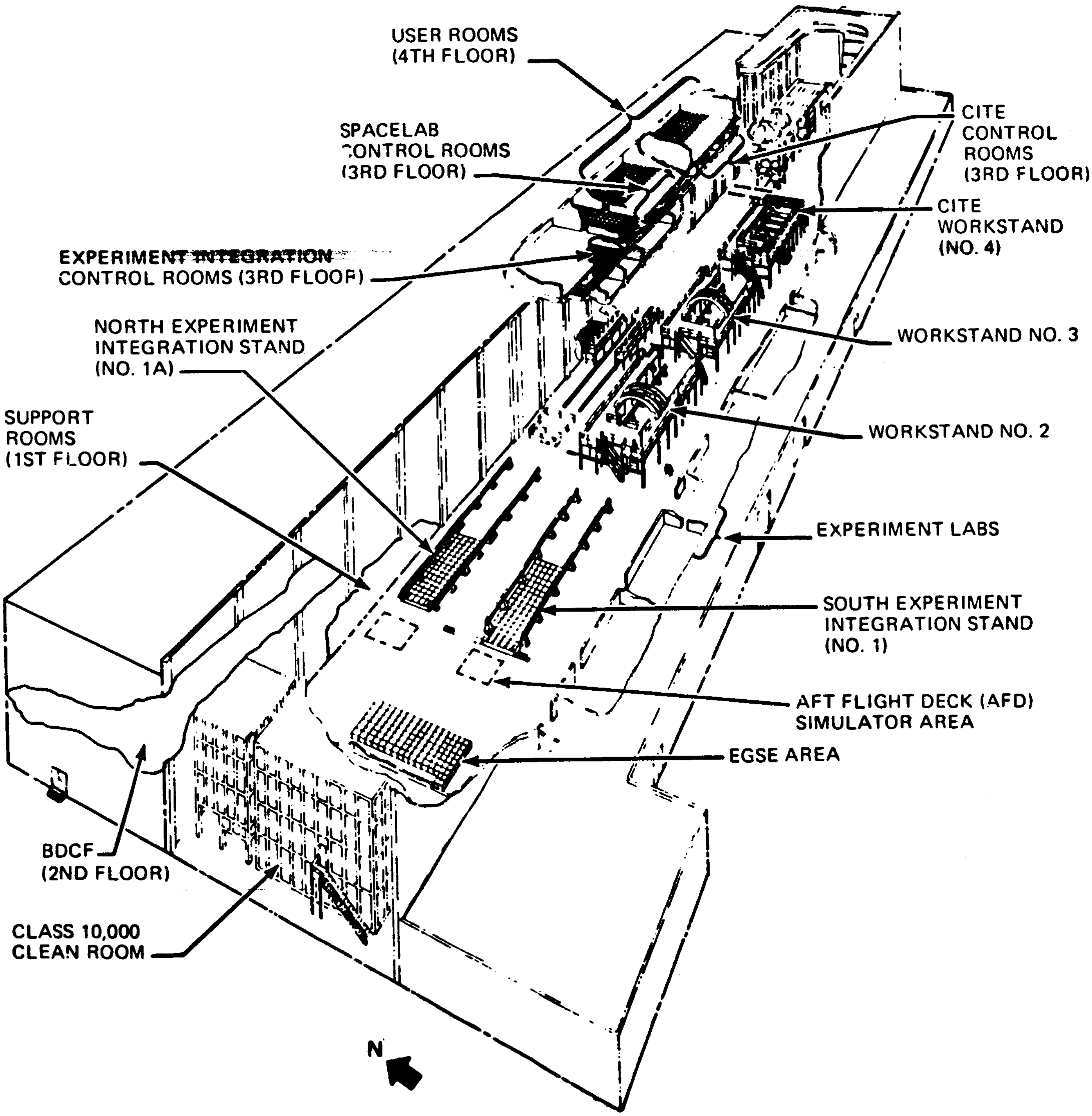


Legend

- A, B, C, D, E, F Storage Bays
- G Utility Closets

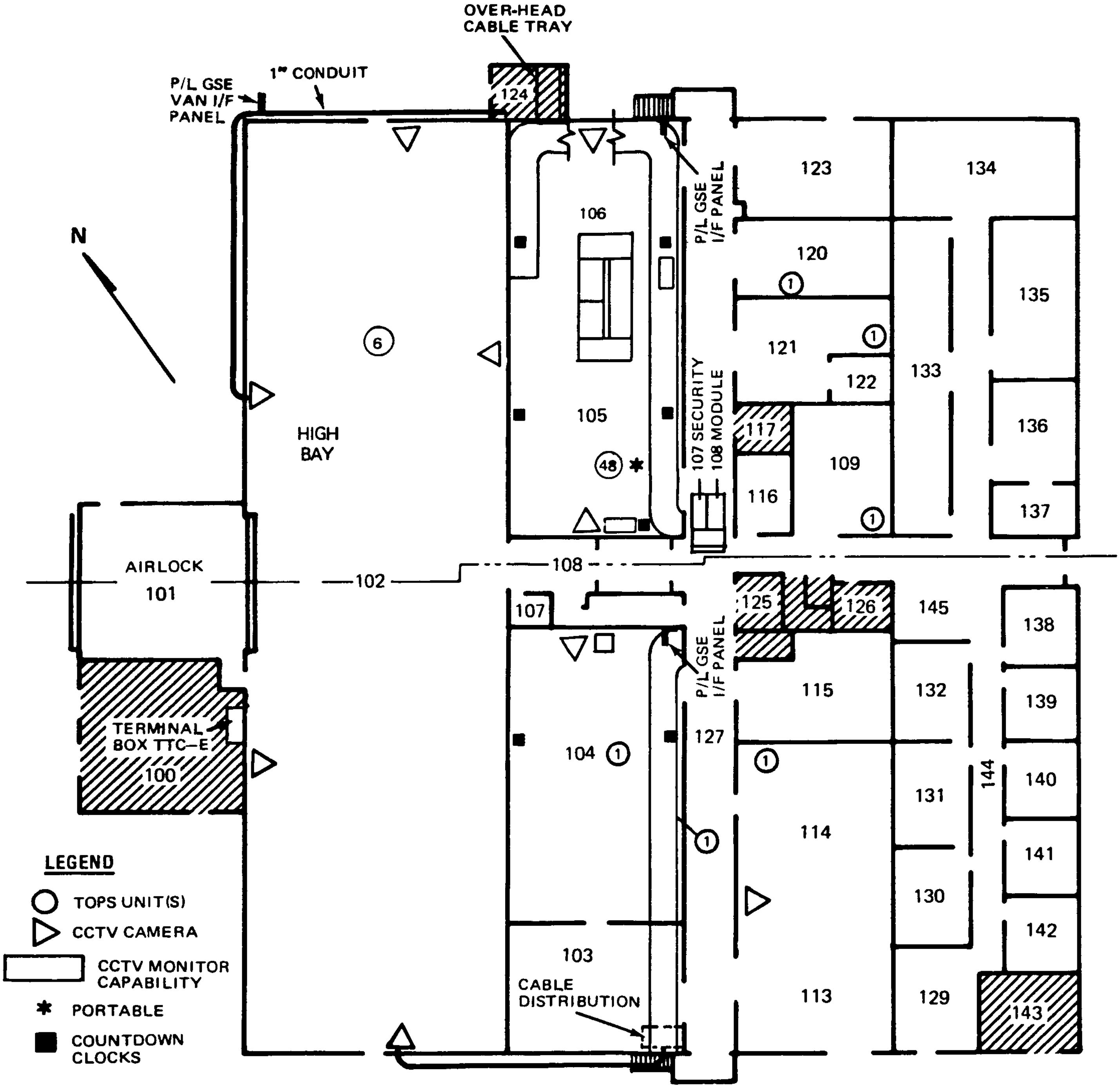
ASTROTECH FACILITIES

Figure 9



USER ROOMS
(4TH FLOOR)
SPACELAB
CONTROL ROOMS
(3RD FLOOR)
CITE
CONTROL
ROOMS
(3RD FLOOR)
CITE
WORKSTAND
(NO. 4)
EXPERIMENT ~~INTEGRATION~~
CONTROL ROOMS (3RD FLOOR)
NORTH EXPERIMENT
INTEGRATION STAND
(NO. 1A)
WORKSTAND NO. 3
SUPPORT
ROOMS
(1ST FLOOR)
WORKSTAND NO. 2
EXPERIMENT LABS
SOUTH EXPERIMENT
INTEGRATION STAND
(NO. 1)
AFT FLIGHT DECK (AFD)
SIMULATOR AREA
EGSE AREA
BDCF
(2ND FLOOR)
CLASS 10,000
CLEAN ROOM
N

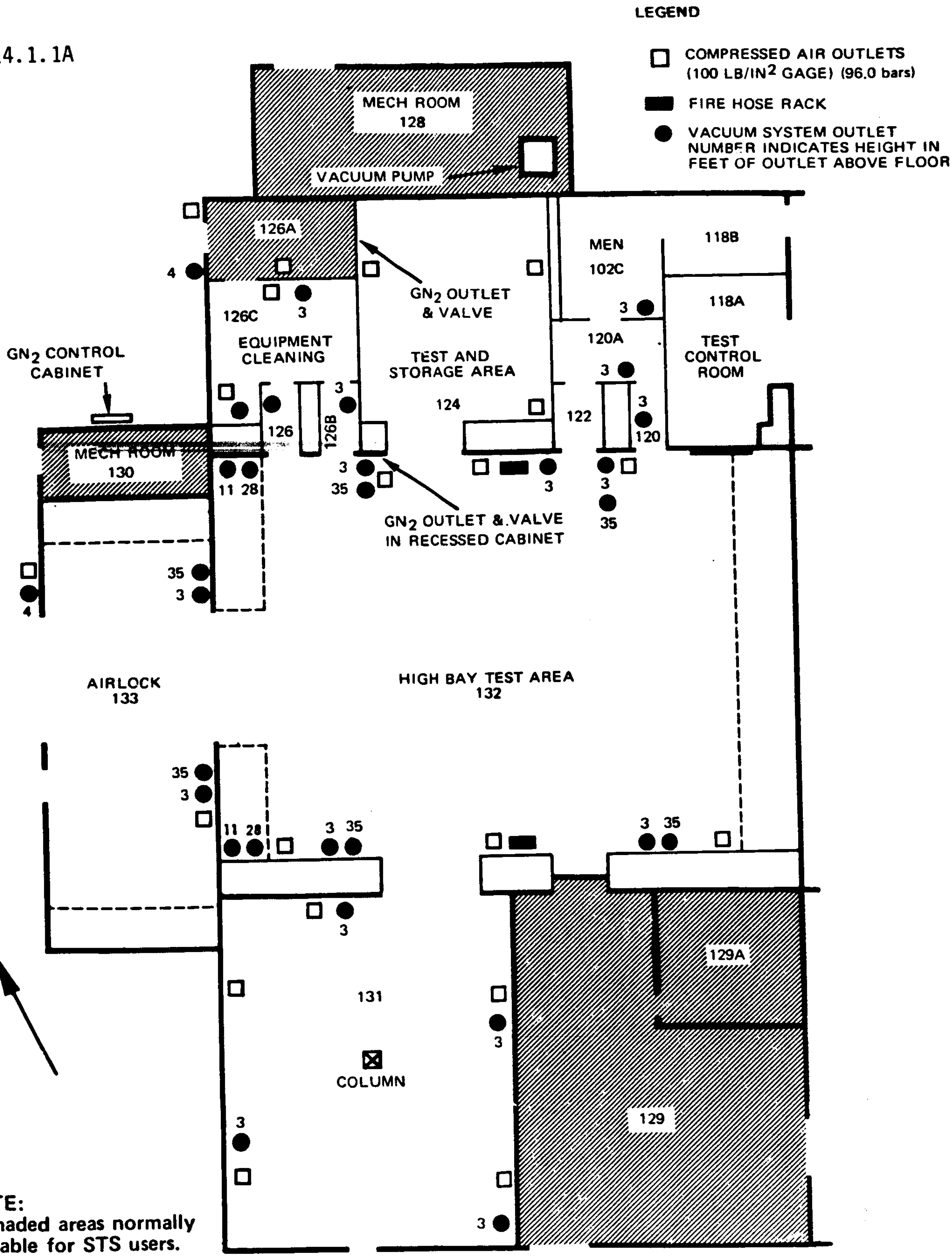
Figure 10



NOTE: Unshaded areas are normally available for STS users.

TOPS AND CCTV LOCATIONS, FIRST FLOOR, BUILDING AO
FIGURE 6–1

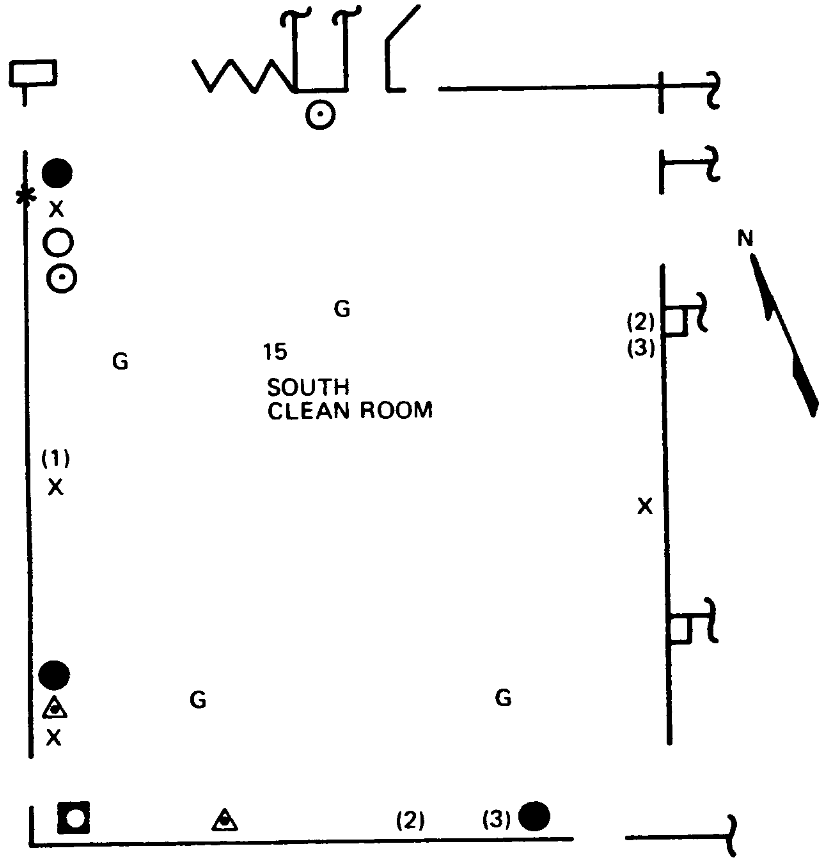
Figure 11
318



NOTE:
Unshaded areas normally available for STS users.

HIGH BAY MECHANICAL SYSTEMS, BUILDING AE
FIGURE 5–2

Figure 12

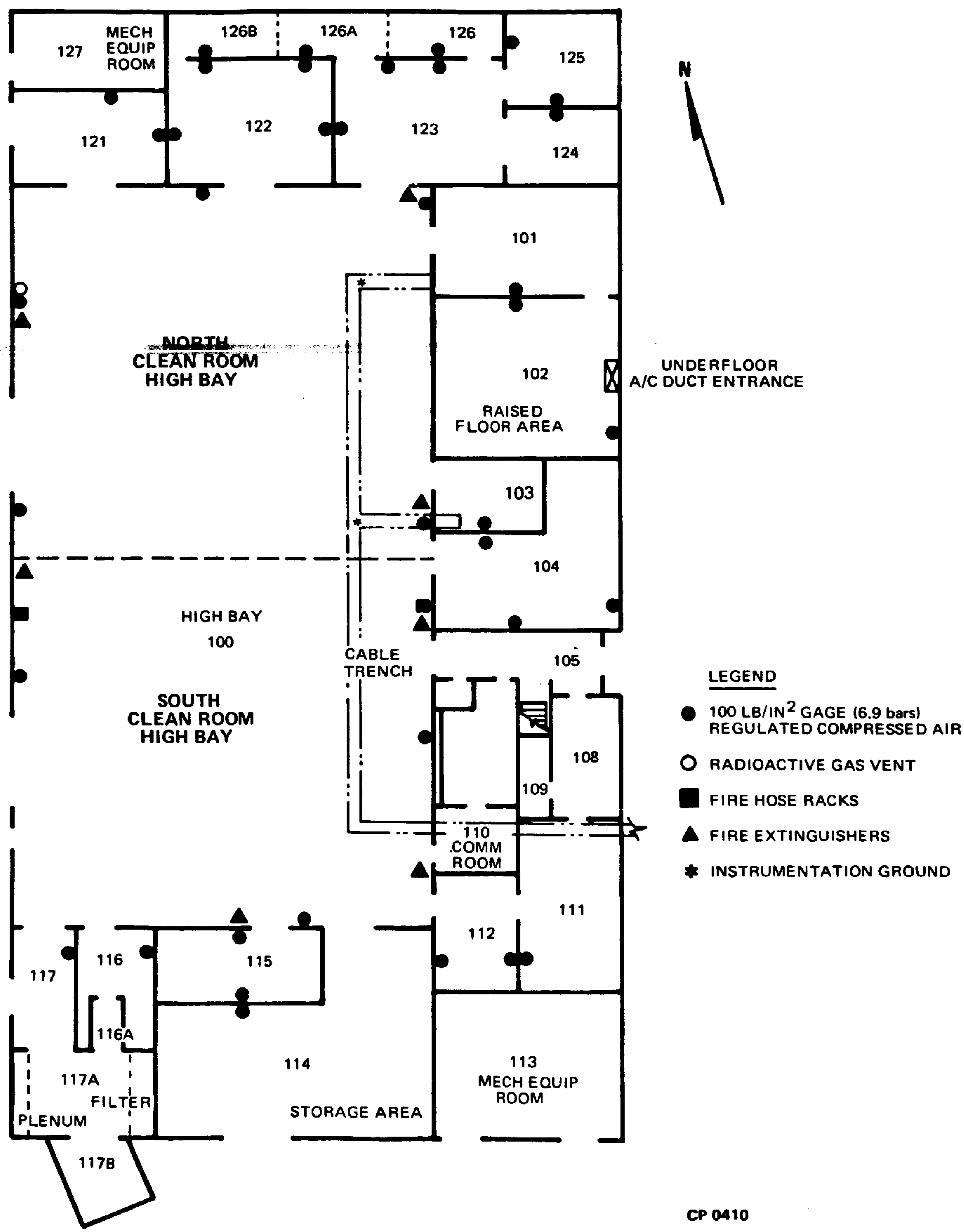


LEGEND:

X	VACUUM
●	GHE AND GN^2 INTERFACE
○	FIRE EXTINGUISHER
◬	COMPRESSED AIR, 100 PSIG (PERFECTING SERVICE CO BRE CO DIV 2F SERIES)
⊙	COMPRESSED AIR, 100 PSIG (HANSEN 3HK)
◘	FIRE HOSE
*	3-INCH HOLE THROUGH WALL
(1)	220-V, 30-A, 1-POLE HUBBLE RECEPTACLE
(2)	110-V, 1-POLE, 3-PHASE, 50-A HUBBLE RECEPTACLE
(3)	8 FLUSH-MOUNTED R&S 3743 RECEPTACLES
G	GROUNDS

MECHANICAL SYSTEMS, SOUTH CLEAN ROOM, HANGAR S
FIGURE 5–1

Figure 13



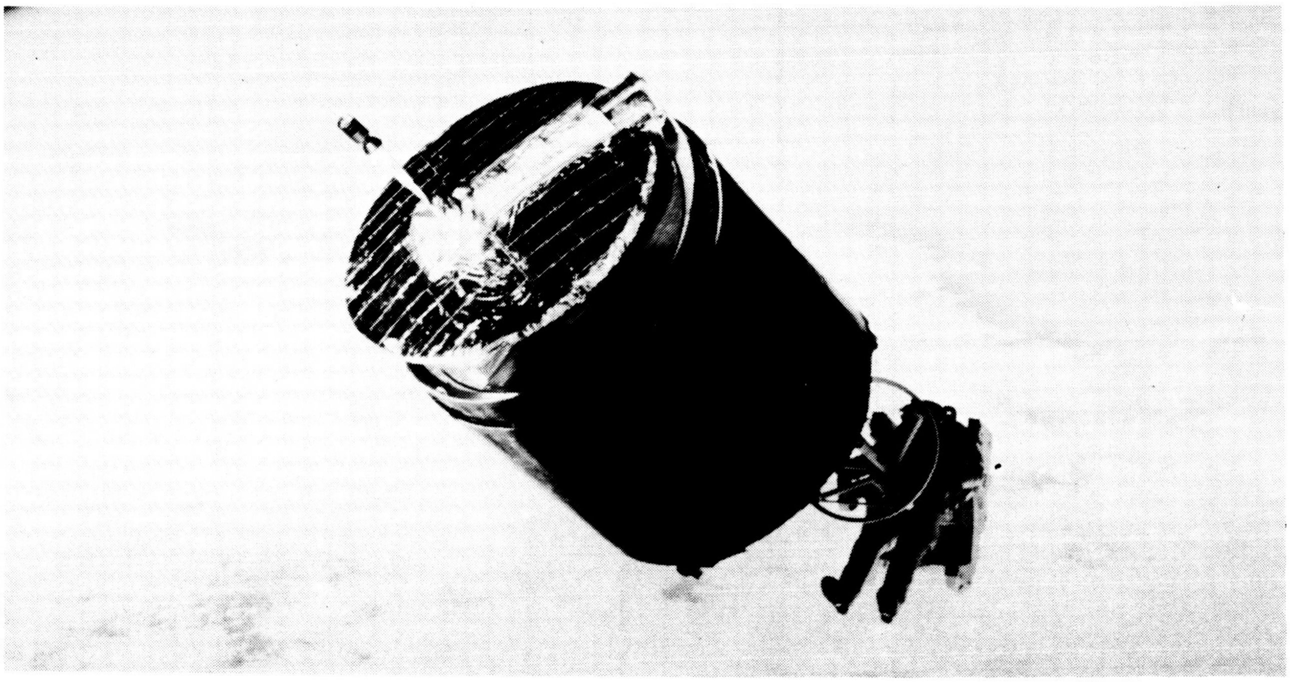
MECHANICAL SYSTEMS, BUILDING AM
FIGURE 5–1

Figure 14

HUGHES

SPACE AND COMMUNICATIONS GROUP

RECOVERING HS 376 SPACECRAFT: FINDING THE HANDLE



J. F. Juraco

For presentation at the
Satellite Services Workshop II

Nov 6-8, 1985
Johnson Space Center

SCG 850610T

Recovering HS 376 Spacecraft: Finding the Handle

J. F. Juraco
HS 376 Recovery Hardware Program Manager

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SCG 850610

ABSTRACT

In February 1984, two HS 376 spacecraft were stranded in useless orbits, unable to achieve their desired synchronous orbits. Almost immediately, teams at Hughes and NASA began efforts to recover the completely healthy spacecraft. Over a period of only 9 months, the spacecraft were maneuvered into a rendezvous orbit, and hardware was designed and built to capture and stow the two "nonrecoverable" vehicles. This document discusses the efforts associated with that recovery mission, which was successfully completed on STS 19 (51-A) in November 1984.

INTRODUCTION

Westar VI and Palapa B2, two of Hughes Aircraft Company's successful series of HS 376 commercial communication satellites, were deployed from the STS bay on 3 and 6 February 1984, respectively. Failures in their perigee kick motors (PKMs) resulted in short burns which left them both in transfer orbits with apogee altitudes of approximately 1200 km, well below their desired synchronous altitudes. Because the spacecraft were in perfect health, in orbits close to those of typical STS missions, and had a large ΔV capability (10 years of hydrazine plus a 1525 meter per second (m/sec) solid apogee motor), recovery of the two spacecraft was considered almost immediately. With the assignment of a potential mission window (originally October 1984), recovery operations planning and design began. Attention focused quickly on two major problems: maneuvering and controlling the spacecraft (in "formation") in low earth orbits, at low spin speeds, with limited command and telemetry visibility; and finding structurally suitable and safe methods of capturing and stowing spacecraft that were not designed to be recovered.

CONSTRAINTS/LIMITATIONS

At first glance, the necessary orbital operations were straightforward - that is, the maneuvers (reorientations, in and out-of-plane ΔV increments, spin speed control, etc) were essentially identical to those performed during a nominal mission. Here they were hampered, however, by limited visibility and the constant constraint to maintain spacecraft attitude within certain bounds for thermal, power, and attitude determination reasons. Another major difference was the requirement for operations at very low spin speeds (0.21 radians per second ultimately) where spacecraft control had never been attempted and dynamic stability had not been characterized. This was exacerbated by the magnetic field and the aerodynamics of low earth orbit, which tended to secularly and periodically precess the spacecraft spin axis and decrease orbit altitude. The final recovery attitude had to be restricted for crew sighting reasons as well.

The capture/berthing operations were totally new. Mating the spacecraft to an adapter for launch is a considerable and tedious task in a 1 g field, where visibility and accessibility are reasonably good. In weightless conditions, with more limited accessibility and an astronaut in a bulky suit with less than desirable maneuverability, the effort could be difficult. The hardware had to be simple to operate and extra strong (to minimize the test and design efforts) and the mating/tiedown visually and unambiguously verifiable. Electrical and thermal interfaces with the STS had to be minimized (even though the spacecraft used these interfaces for launch), and finally, the entire effort had to be safe for both crew and spacecraft.

SPACECRAFT DESCRIPTION

An exploded view of a typical HS 376 spacecraft is provided in Figure 1. The spacecraft is designed as a dual spin vehicle in its operational configuration, with a rotor (spinning section) and a platform (despun section). The rotor contains the housekeeping functions - specifically the power, propulsion, and attitude control subsystems - while the platform supports the communications payload, including antennas. The central thrust tube is the structural heart of the spacecraft. It houses the apogee motor and supports all other spacecraft structure. At the aft end of the thrust tube is the separation ring (perigee motor interface). The attitude control subsystem includes redundant dual-slit sun sensors and earth sensors, both of which scan via satellite rotation (for attitude determination), and a linear accelerometer mounted parallel to and offset from the spin axis (for nutation control). The propulsion subsystem employs four (nominally) 22 newton thrusters operated in a blowdown mode (thrust proportional to tank pressure). Thrust decays to about 4.5 newtons near end of mission. Two thrusters fire axially to provide axial ΔV, spin axis precession, and limited spin speed control. The other two fire radially to produce ΔV and primary spin speed control (one is canted for spinup, and the other for spindown). In the on-station configuration, the radial thrusters fire through the spacecraft center of mass, but in transfer orbit radial ΔVs are considerably offset, producing transverse torques up to 60% of their spin torque component. Because they fire through solar panel cutouts, duty cycle and pulse width constraints of 12% and 1 second must be observed to avoid solar panel damage due to plume heating.

Ground handling is performed by interfacing with hard points on the spun shelf (above structural ties to the thrust tube), through thruster and earth sensor cutouts in the solar panels. These hard point interfaces are removed and the cutouts fitted with closeouts before launch.

During launch and transfer orbit (and during recovery operations), the platform and rotor are locked together as a single rigid body, and the aft solar panel is stowed and locked to the forward. The reflector is stowed and tied to the feed horn assembly. Only the omni antenna is deployed (see Figure 2).

For the most part, all the readily accessible portions of the spacecraft are quite fragile. The thrust tube separation ring is the only point at which the spacecraft can be held for return. Its accessibility was limited, however, because both the solar panels and the apogee motor nozzle extend beyond the separation plane. At the forward end, only the spacecraft bumper bracket (which acts as an antenna stop) and the feed

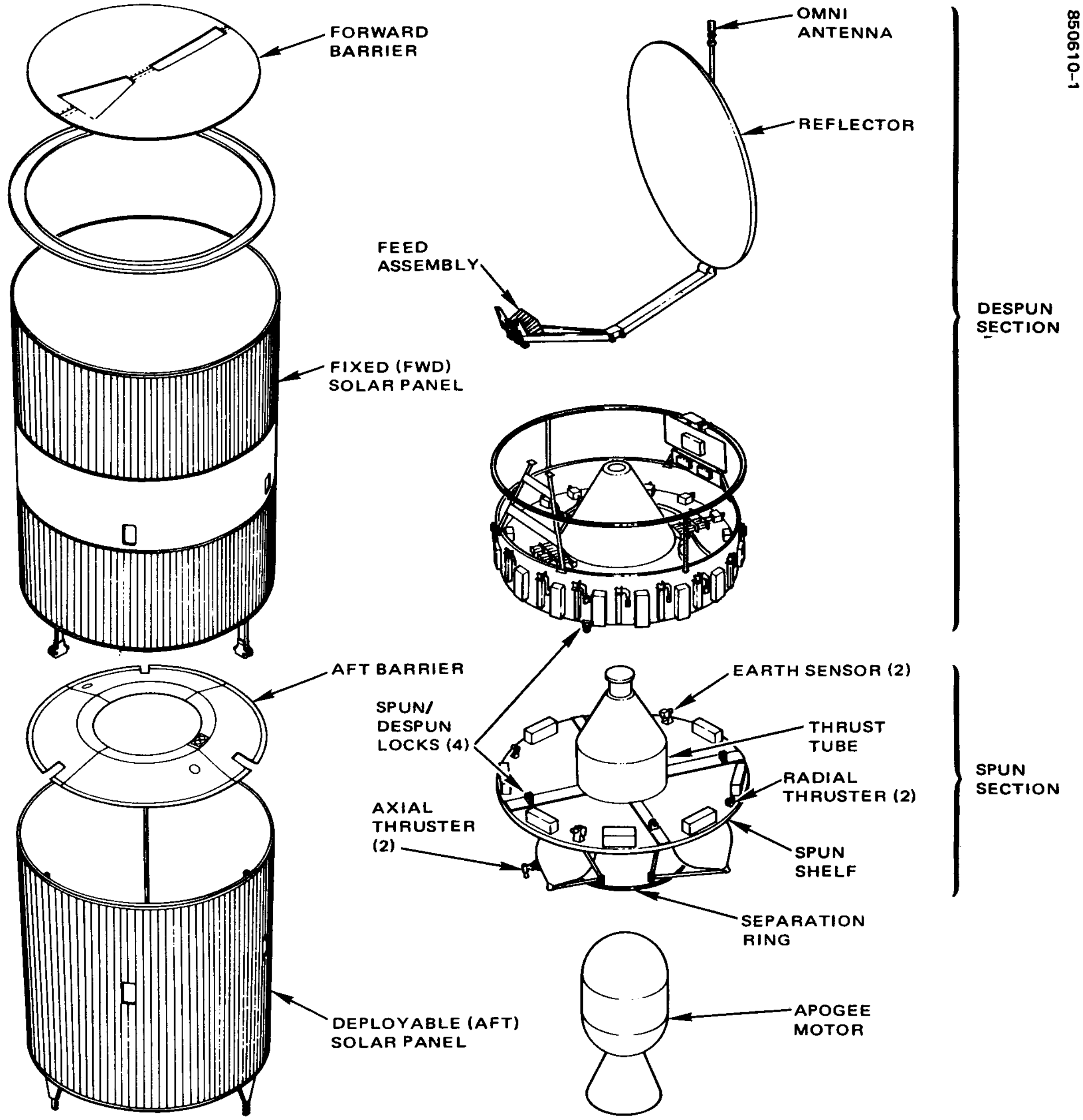

FIGURE 1. HS 376 SPACECRAFT EXPLODED VIEW

assembly afforded any handling capabilities. These were the areas upon which the capture and stowage planning focused.

Table 1 summarizes the physical characteristics of the recovered spacecraft.

ORBITAL OPERATIONS

The PKMs were supposed to increase the satellite's velocity of about 7625 m/sec by 2440 m/sec, but the short burn supplied an increment of only about 244 m/sec. The misfires also left the inclinations of the two orbits different and diverging.

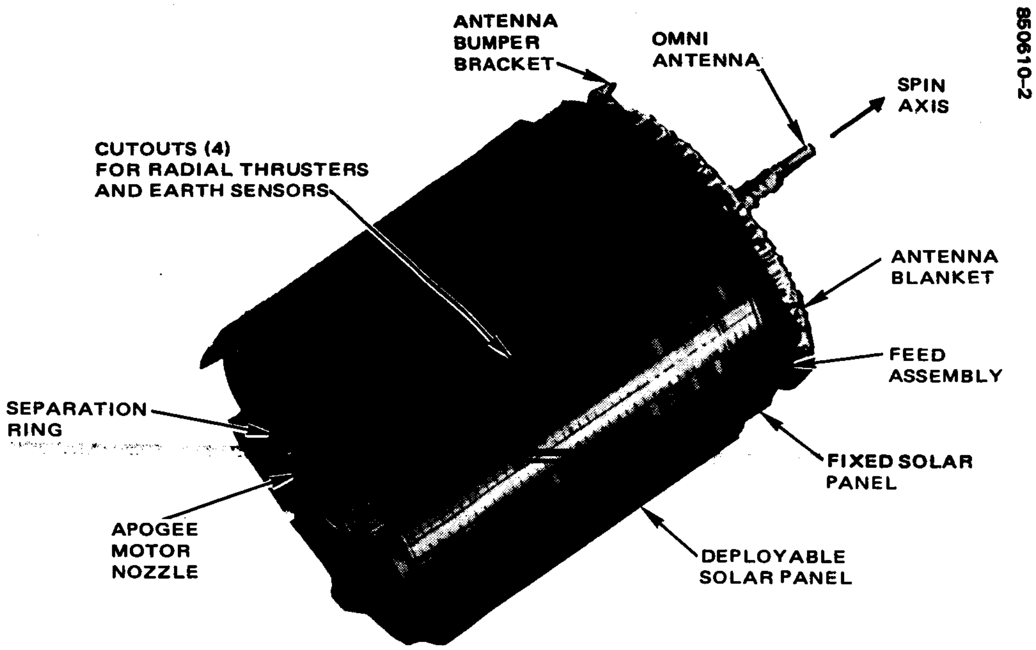

FIGURE 2. HS 376 TRANSFER ORBIT CONFIGURATION

TABLE 1. PHYSICAL CHARACTERISTICS

	Westar VI	Palapa B2
Mass after PKM separation, kg	1102	1228
Hydrazine capacity, kg	147	210
Spacecraft dry weight, kg	472	513
Spacecraft returned weight, kg	490	547
Spin/transverse inertia ratio at rendezvous	1.05	1.06
On station length, m	6.5	6.8
Launch length, m	2.8	2.8
Diameter, m	2.1	2.1
Nominal spin speed, rad/sec	5.2	5.2

The first major orbital operation was the firing of the apogee motors about 3 months after the original mission. This essentially out-of-plane firing made the spacecraft lighter (allowing lighter recovery hardware), removed the apogee motor as a safety hazard, and raised the perigees out of the atmospheric drag for the considerable time remaining until the recovery mission. After the firings, both spacecraft were in essentially circular orbits at altitudes of 1100 to 1500 km and at 28.5^{o} inclinations with nodes 18^{o} apart and converging at a rate of 0.25^{o}/day (see Figure 3). The orbits were

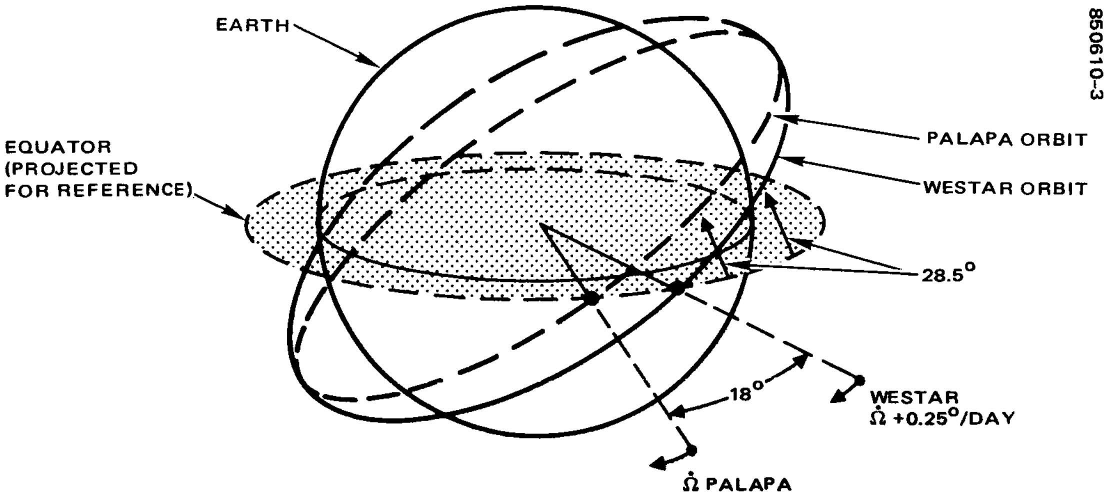

FIGURE 3. CONVERGING WESTAR VI AND PALAPA B2 ORBITS

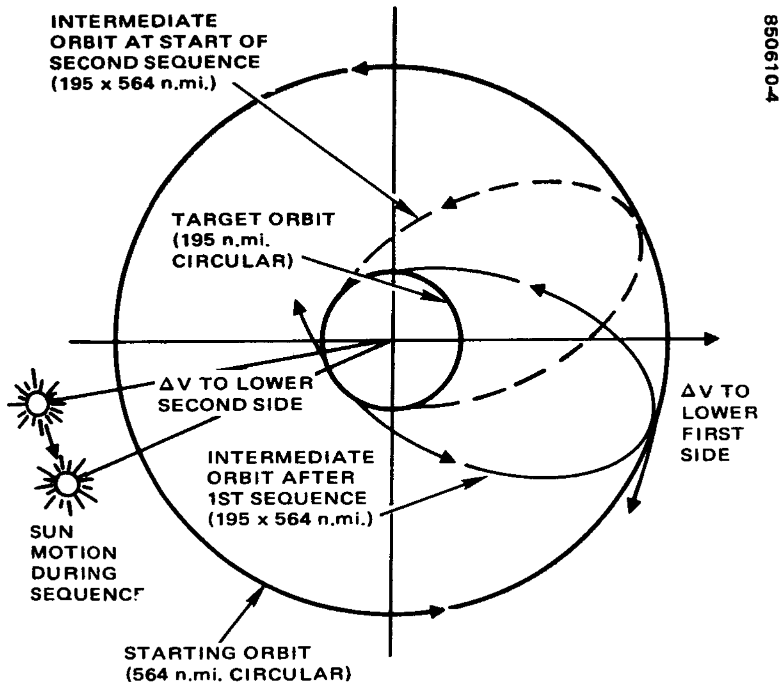

FIGURE 4. TRANSFER OF WESTAR VI/PALAPA B2
TO SHUTTLE RENEZVOUS ORBIT

synchronized in August after a sequence of 33 manuevers. The spacecraft were also spun down to 2.1 rad/sec at this time.

The most difficult operations were performed in the 3 weeks before the recovery mission when over 100 maneuvers were performed on each spacecraft to place them 10° apart (2.5 minutes apart as they passed overhead) in a circular orbit at an altitude of 360 km. This sequence is shown in Figure 4. The spacecraft were also spun down to 0.21 rad/sec during this period.

Spindown was difficult due to the greatly reduced time available for attitude determination (although the low spin speed also meant there were less data to analyze), the sensitivity of the spacecraft to transverse torques at low spin speed due to thruster offset and the reduced angular momentum, and the fact that the automatic attitude data processing hardware on the spacecraft ceased functioning below 2.1 rad/sec (the onboard automatic nutation damping function ceased as well). Ironically, it became necessary to reduce by hand the raw sensor pulses which were telemetered from the spacecraft.

On the other hand, the slow data accumulation, the short reaction time, and the spacecraft dynamics called for an implementation of ground station automation to produce alternate thruster firings in rapid succession, which applied the correct amount of attitude precession or orbit ΔV without creating destabilizing spacecraft nutation. While it was originally planned to spin down to 0.05 rad/sec, crew simulations showed that 0.21 rad/sec (or higher) was acceptable, as long as spacecraft attitude was stable with minimum coning. The two spacecraft were stable, but the passive damping time constant at low tank fill fractions was about 12 hours.

Final spin speed was attained approximately 1 week before rendezvous to allow time to calibrate aerodynamic and magnetic moment disturbances so that an attitude suitable for visual sighting could be set up well before approach by an astronaut. This required that the reflective antenna blanket be in sunlight as the STS came over the horizon. Finally, about 24 hours before rendezvous, all commandable spacecraft electronics were turned off, and all hydrazine latch valves were commanded closed.

CAPTURE AND BERTHING

The original plan required an astronaut in a manned maneuvering unit (MMU) to attach a grapple fixture to the spinning spacecraft bumper bracket and to subsequently null all spacecraft rates so the shuttle could safely approach and grapple. The remote manipulator system (RMS) would then lower the spacecraft onto a clamp device built on a cradle. The astronaut would tighten the clamp to the required preload for return. While this plan required minimal new hardware and only one astronaut to perform the entire operation, it was discarded for a number of reasons. First, the bumper bracket was small, well blanketed, relatively hard to get at (any structurally meaningful grasp of the bracket required some movement into the spacecraft), and offset from the spin axis. Later analyses also showed that it was not capable of handling the runaway RMS loads. Furthermore, if the astronaut were to hit the spacecraft and not complete the attachment, the resultant spacecraft motion would almost certainly stop at least that particular attempt. Finally, because of the size and location of the separation ring, RMS accuracy and overall sighting limitations made a proper berthing of the spacecraft unlikely.

Several plans later, it was decided that an astronaut in an MMU would fly to the spacecraft with an apogee motor capture device - a stinger - in front of the MMU (Figure 5). Using the nozzle as a made-to-order target, he would insert the stinger into the burned out motor and release four toggle fingers which would hold him to the spacecraft (Figure 6). A jackscrew extending through the stinger would allow him to tighten a circular brace against the separation ring, forming a rigid connection of spacecraft, stinger, and MMU. Once the spacecraft was under control, the RMS would grab the grapple fixture on the MMU and position the spacecraft such that its forward

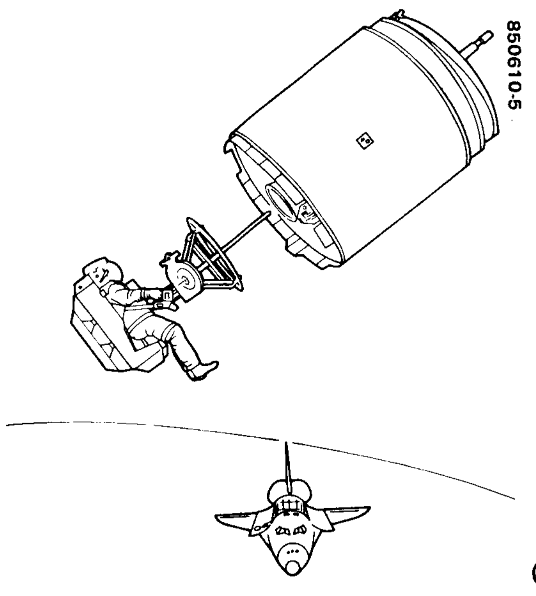

FIGURE 5. STINGER PENETRATION AND SATELLITE CAPTURE

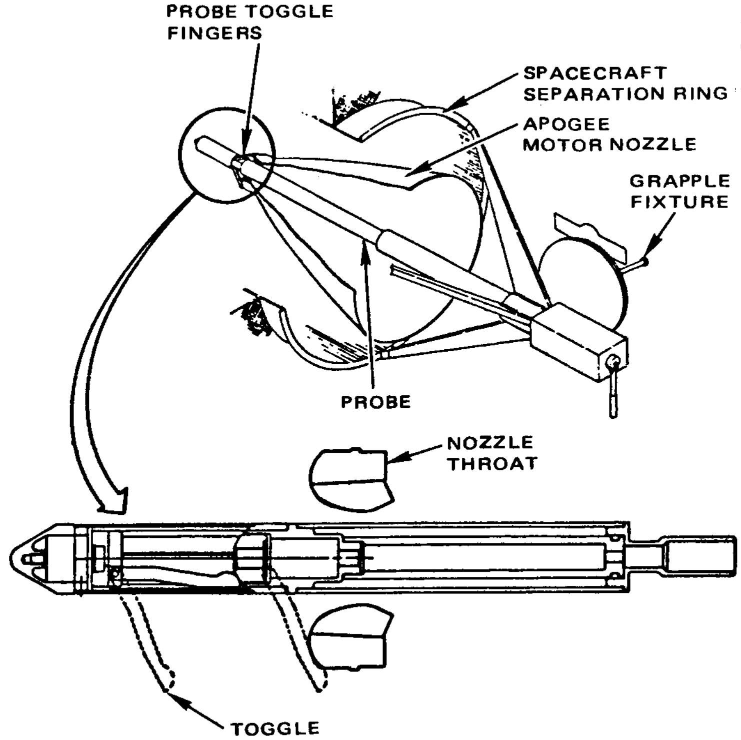

FIGURE 6. STINGER INSERTED IN APOGEE MOTOR NOZZLE

end was over the payload bay. A second astronaut would then attach an antenna bridge structure (ABS) to the spacecraft forward end (after the omni was cut off), picking up the bumper bracket and the antenna feed assembly (i.e., bridging the antenna) as shown in Figure 7. The RMS would transfer to the ABS grapple fixture, after which time the MMU would disengage. The MMU had to remain attached during ABS installation so spacecraft control could be maintained while the RMS moved to the ABS. The ABS was required in order to free the separation ring for eventual adapter installation.

The exultation that followed the quick stinger capture was quickly dampened when ABS installation failed because of a short protrusion (approximately 0.3 cm) on the antenna feed assembly. The acronym ABS soon became the "astronaut" bridge structure when the astronaut, per the contingency procedures, held the spacecraft via the four-bar linkage along the back of the reflector. Standing in a mobile foot restraint (MFR) attached to the STS sill, he positioned the spacecraft aft end over the bay. The first astronaut, having disengaged and stowed the MMU/stinger, then raised a 250 kg adapter to contact the separation ring (see Figure 8). Spring loaded latches on the adapter loosely captured the separation ring, freeing the astronaut's hands. At the top of the adapter were nine clamp shoes, each connected to the bottom, for easy access, via a jackscrew arrangement. The shoes were tightened, in a carefully defined sequence, via a torque wrench. A passive indicator on each drive train provided verification of proper preload. Once the adapter was installed, the astronauts manually positioned the spacecraft-adapter combination into three payload retention latch assemblies (PRLA) on the cradle platform. The stowed spacecraft is illlustrated in Figure 9.

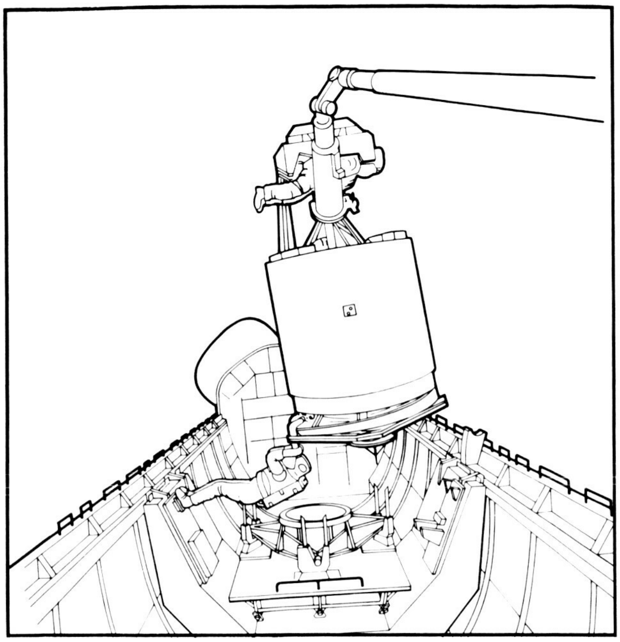

FIGURE 7. ANTENNA BRIDGE STRUCTURE INSTALLATION

FIGURE 8. ASTRONAUT HOLDING SPACECRAFT OVER BAY FOR ADAPTER INSTALLATION

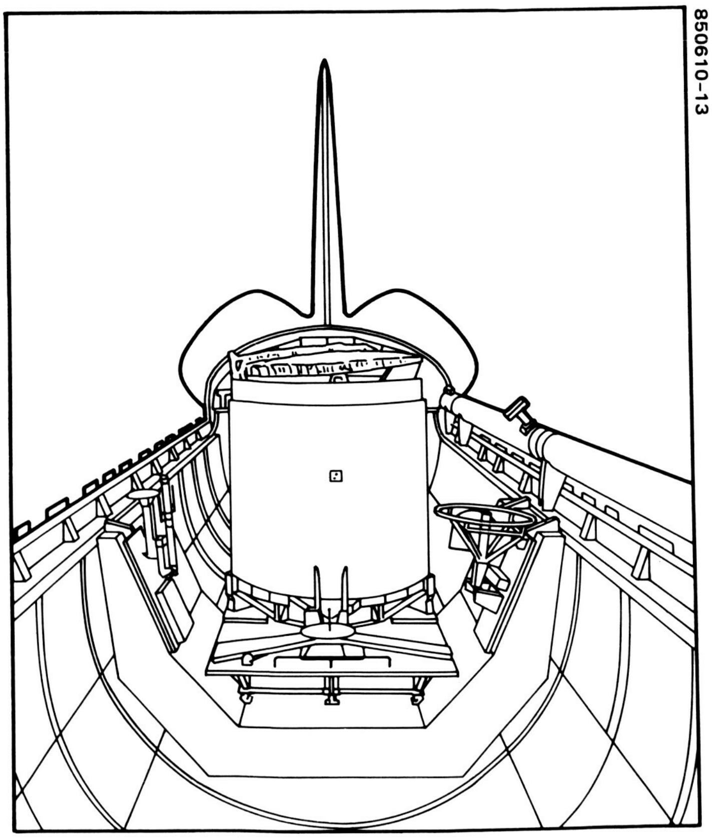

FIGURE 9. SPACECRAFT STOWED IN BAY

FIGURE 10. USE OF RMS TO SECURE SPACECRAFT/MMU

The ABS had an interference problem with only the first spacecraft (Palapa B2). Exhaustive ground checks showed that Westar VI did not have the same protrusion. However, the manual operations, once rehearsed, were preferable so capture of the second spacecraft 2 days later proceeded without the ABS. The only major difference with respect to Palapa B2 operations was the use of the MFR at the end of the RMS (rather than on the sill) as shown in Figure 10. This made it easier to hold the spacecraft over the bay for adapter installation and was required in any case due to the more restricted work area in the bay caused by stowage of the first spacecraft. Also, the omni antenna was not cut off until after the spacecraft was stowed, thereby making available another "handle". (Omni removal was required for bay door closure.) The capture/berthing went exactly as planned.

RECOVERY HARDWARE

The primary recovery hardware is shown in Figure 11. A spacelab pallet was adapted for use as the main stowage frame. A 2.1 by 3.1 meter platform was fitted to the pallet to raise the mounting surface so the spacecraft solar panels would not contact the sloped sides of the pallet. On the platform were mounted the three PRLAs which held the adapter during launch and the adapter/spacecraft during reentry. The pallet/platform also held the stinger, the ABS, and other miscellaneous tools.

The adapter is shown in more detail in Figure 12. Nine guide rails at the top of the adapter (separation ring interface) helped guide the adapter onto the spacecraft. Three of the guide rails held the spring loaded latches (soft dock clamps), which allowed loose capture of the spacecraft and freed the astronaut for other tasks. If the adapter was misaligned with the spacecraft during the mating process, the astronaut could quickly release these latches and try again. Alternating with the guide rails were the nine clamp shoes. All clamp devices were controllable from the bottom of the adapter. The adapter had three legs, with a trunnion at the end of each leg. The legs were

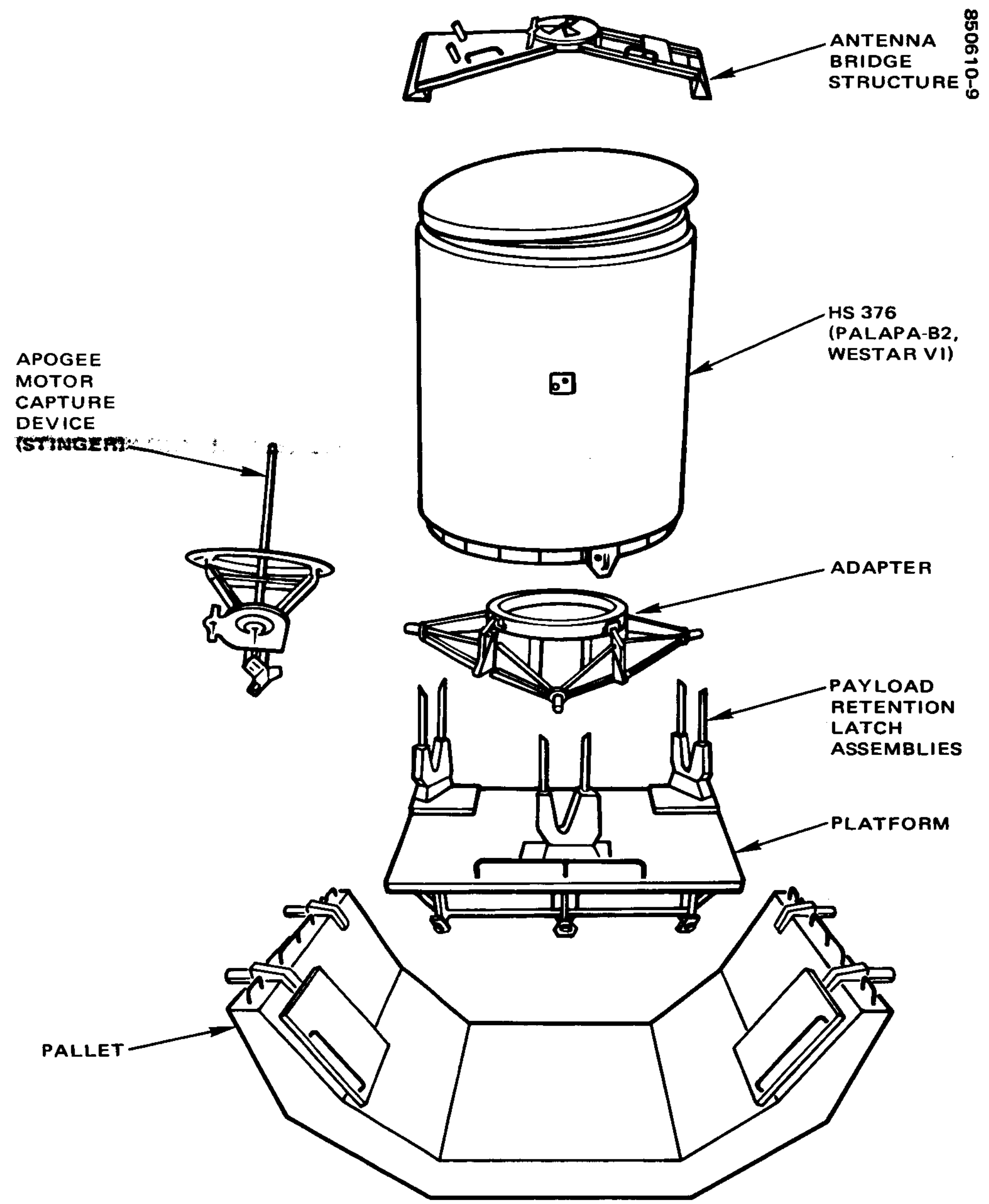

FIGURE 11. RECOVERY MISSION HARDWARE

required to provide an attach point beyond the perimeter of the spacecraft to allow for visual verification of tiedown. The motor-driven PRLAs were controlled from the cabin.

The functional and design requirements of the adapter belied its apparent simplicity. Most importantly, it had to work; there were no alternatives or contingency procedures for holding the spacecraft. At a detailed level, the separation ring is quite flexible so a proper combination of the number of contact points (shoes) and contact point preload was needed to hold the spacecraft during worst case reentry loads. Use of nine shoes was the best overall solution, six shoes being adequate if each failed shoe was flanked by two good ones. Close tolerances were also required. The ring diameter manufacturing tolerances were ±0.025 cm, and when possible temperature extremes were

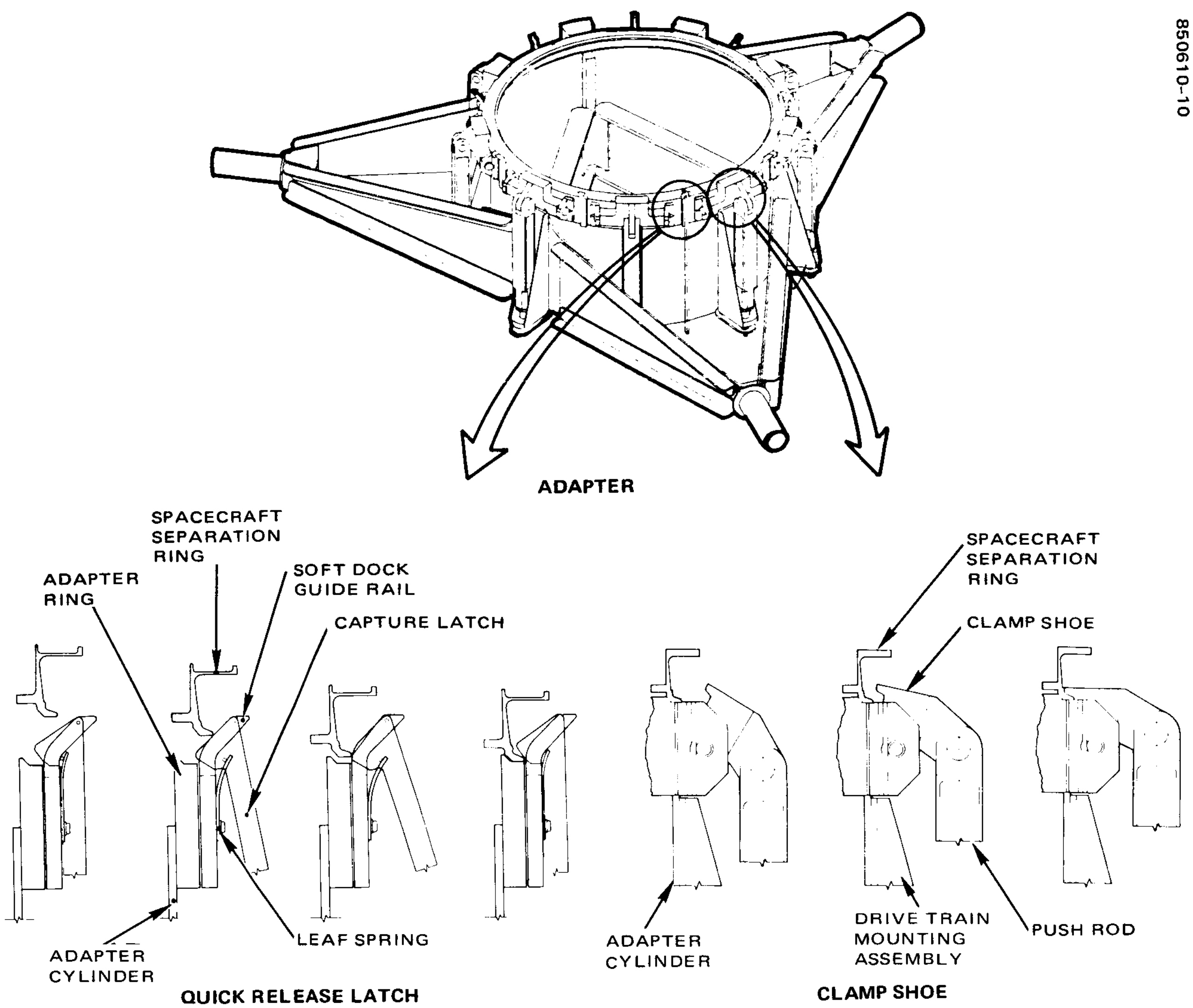

FIGURE 12. RECOVERY ADAPTER

considered (e.g., cold spacecraft, hot adapter), proper mating would not take place. The clamp shoes, therefore, had to force both the horizontal and shear lip surfaces together, regardless of the temperature mismatch. Analyses later showed that the two items would attain thermal equilibrium before the torqueing sequence could be completed. Finally, because of frictional (and consequently required torque) variations between drive trains, a foolproof indication of proper preload was required (other than having the astronaut torque each drive train to a different value). Relying on the fact that strain in each drive train repeats with preload (while torque may not), a simple plate (displacement indicator) was attached to each drive shaft (see Figure 13). When the indicator was flush with the bottom of the drive train mounting foot, torqueing could stop. Therefore, specific torque values never became a mission consideration although the actual variations (45 to 80 N-m) made for some strenuous work on the part of the astronauts.

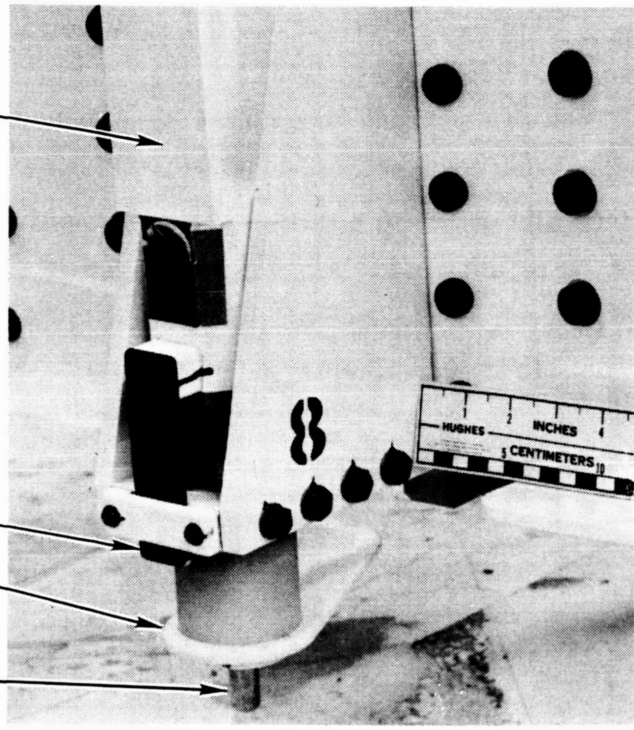

FIGURE13. DISPLACEMENT INDICATOR

MISCELLANEOUS

Another primary consideration was the postcapture, in-bay thermal environment of the stowed spacecraft, especially while the shuttle was chasing the second spacecraft. The concern here was the possible repeated freezing and rethawing of hydrazine which might lead to fuel line rupture, fuel leakage, and contamination of the crew as they worked in the bay. Extensive analyses were performed to define a flight profile which guaranteed that fuel temperatures would not fall below freezing, ultimately removing this from the worry list.

SUMMARY/CONCLUSION

The successful retrieval of two HS 376 spacecraft is described. A brief description is provided of the spacecraft, the attitude control considerations, the orbital operations, and the capture/berthing hardware and procedures.

The entire recovery mission, because of its accomplishments in a very short time, provided a clear demonstration of the degree of sophistication and familiarity which industry and NASA have achieved in the area of space design and development and in space itself.

EVOLUTION FROM EVA TOWARD ROBOTICS FOR SATELLITE SERVICING

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This paper addresses questions related to conceivable paths for evolution from labor intensive EVA for satellite servicing and repair toward labor saving robotics and also looks at the effect that different paths will have on the rate of progress toward robotics.

Satellite servicing and repair became a demonstrated function in April of 1984 when the crew of STS Mission 41-C successfully repaired the Solar Maximum Satellite. After failing to lock the MMU to the rotating and undulating Solar Max, the operations were successfully modified to provide capture of the satellite with the RMS and subsequent EVA replacement of the faulty attitude control module. (Reference 1)

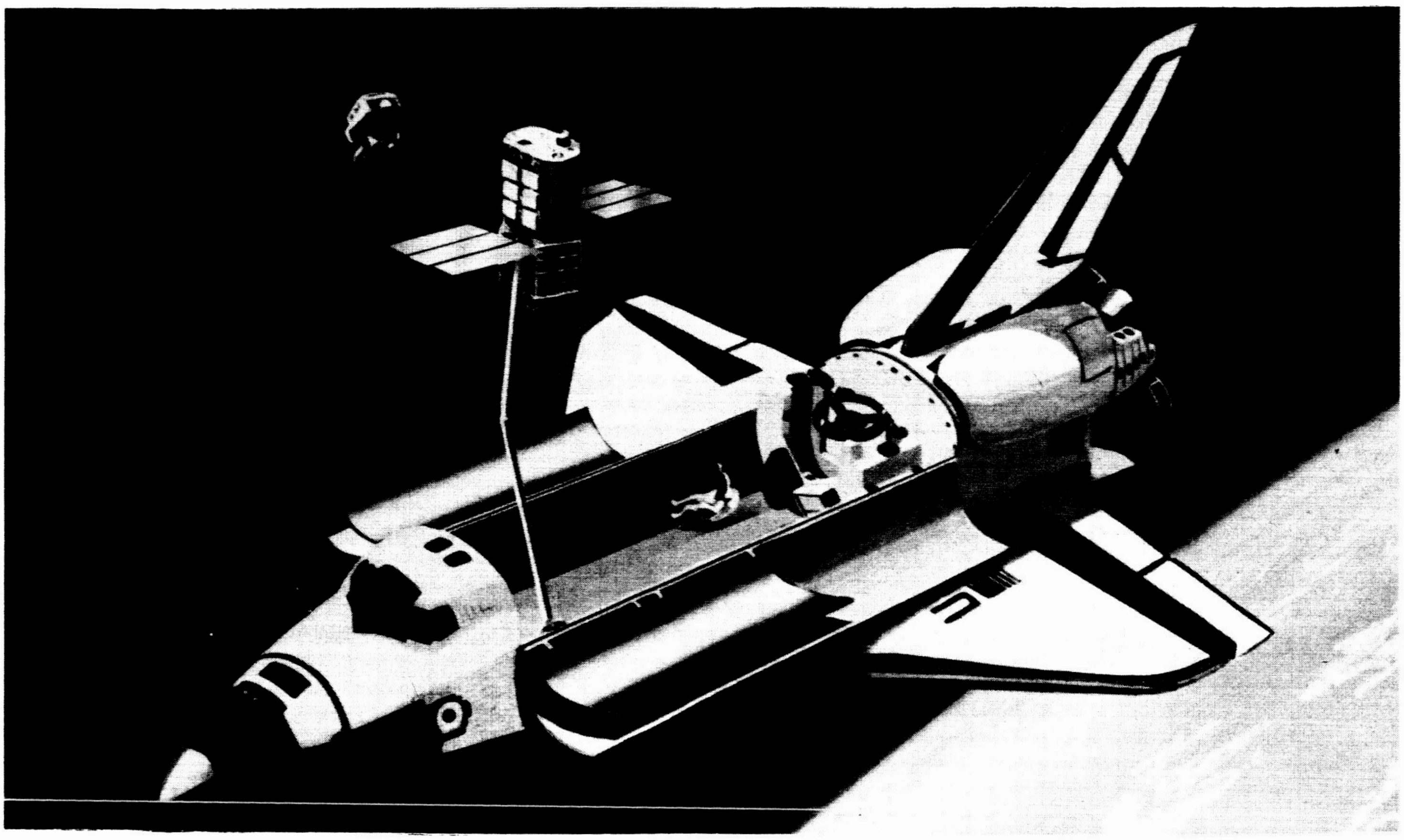


FIGURE 1

More recently a successfully coordinated EVA, RMS and Shuttle operation was conducted on STS mission 51-I to capture the Leasat F3, effect a bypass of its failed sequencer and redeploy the satellite with a 3 RPM spin rate. (Reference 2) This activity was correographed to involve the cooperative efforts of two EVA astronauts, movements of the RMS and shuttle orbiter position and attitude adjustments. As effective as the operation eventually was, it consumed over 10 hours of the attention of four astronauts and also occupied the STS for a period of time beyond those 10 hours. For the day to day operations expected with Space Station satellite servicing and repair, such time and resource consuming missions would soon become unacceptable.

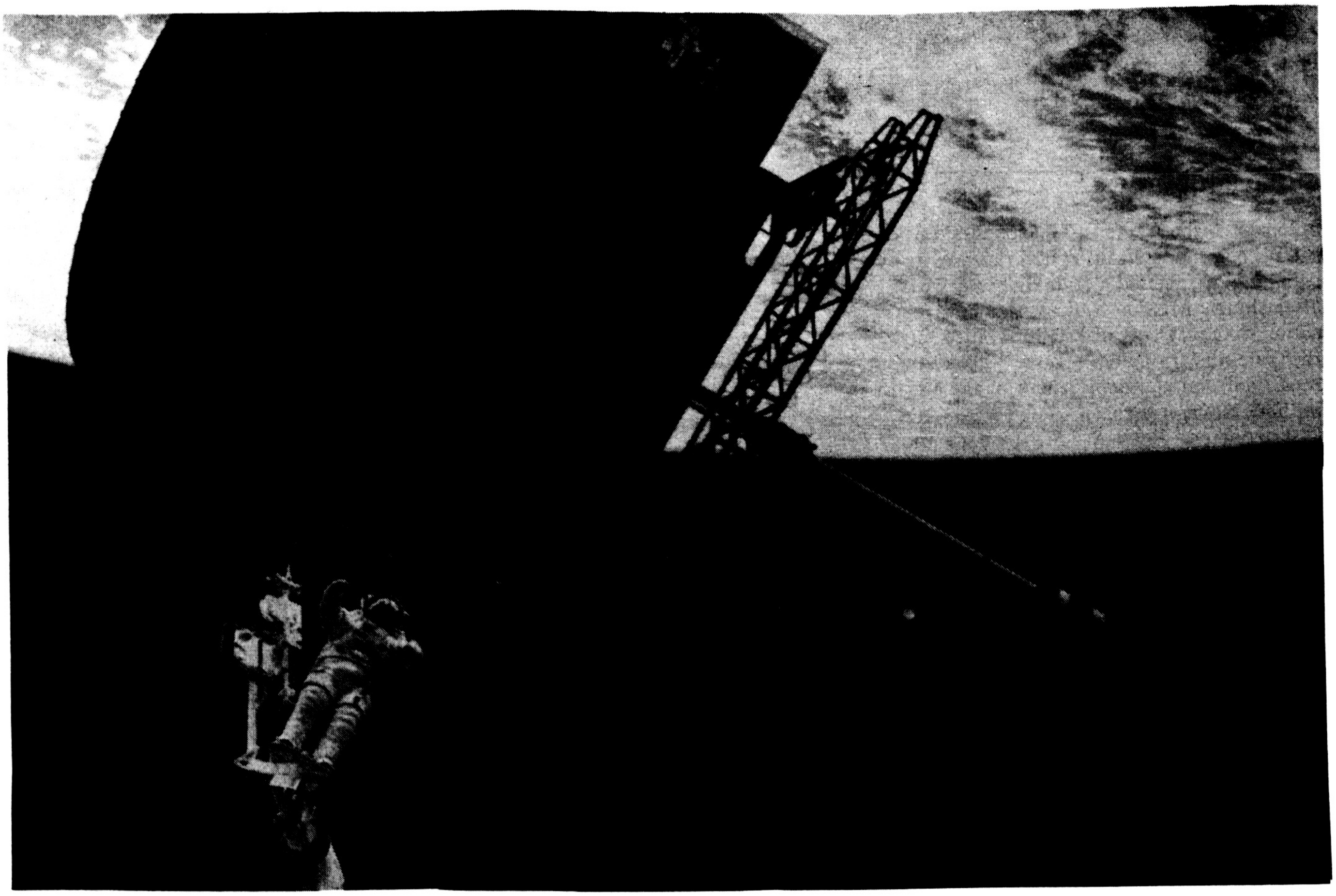


FIGURE 2

In both of the STS missions sited above the adaptability of humans at the task site was important to the success of the operations. In the case of mission 41-C the adjustments after the docking pin failure involved human observations and decision making both on-orbit and on the ground. For mission 51-I the malfunction of the RMS arm, so that only single joint operations were possible, required adjustments which were worked out by humans on the ground and were accomplished by Astronaut Lounge in space. Also on 51-I the shuttle altitude control disturbances requiring a switch to free drift were detected based on observations by EVA astronauts Van Hoften and Fisher. This need to deal with the unforeseen is one of the strongest challenges in the use of automation and robotics in space.

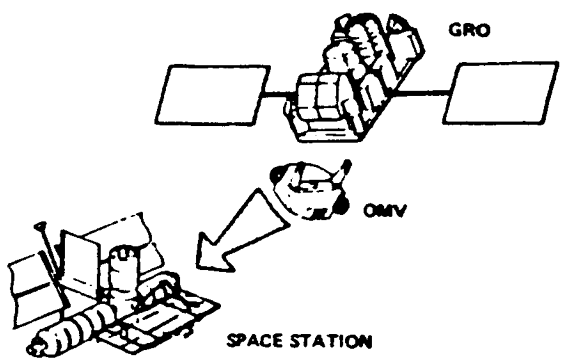
On the other side of the coin, the cost of human time on-orbit is one of the strongest arguments for the use of automation and robotics on the Space Station. It is estimated in various interpretations of the Langley Mission Data Base for the Space Station that satellite servicing will require from 850 to 960 (reference 3) EVA hours per year initially and that is expected to increase within a few years after IOC. Based on 18 hours per week for each of the four EVA crew members, which is the spec limit for EVA imposed by the preliminary phase B system requirements, the total Space Station EVA time is 3744 hours per year. This means that up to 25 percent of the total EVA time will be devoted to satellite servicing. The utility of the Space Station for expanded missions demands that crew time be used productively and the application of automation and robotics to a heavy time user such as satellite servicing can significantly improve that productivity.

Suited EVA is a particularly inefficient way to use human time because of the overhead for donning and doffing and with current suit procedures the 40 minute oxygen pre-breathing prior to each EVA. In addition human functioning on EVA is impaired by the lack of dexterity in the gloves and the restrictions in mobility imposed by the pressurized suit. Those attributes of human capability which allow response to unforeseen events are associated with senses which are also impaired on EVA because touch and sight are constrained by the pressurized suit. Other sensations which allow unique human perception of the task environment such as sound, smell, or feeling on the skin are not available in space or are not transmitted through the suit to the astronaut. For these reasons EVA is a good place to look for applications of automation and robotics. Since satellite servicing is a major user of EVA for the Space Station we need to investigate how automation and robotics can reduce the astronaut time involved with those tasks.

Typical satellite servicing missions are discussed in TRW's report to the NASA Advanced Technology Advisory Committee Study on Space Station Automation and

Robotics (Reference 4). In this report four reference mission scenarios are described covering 1) servicing a Gamma Ray Observatory at the Space Station, 2) servicing at a free-flying materials processing facility, 3) servicing a payload or subsystem attached to the exterior of the Space Station and 4) servicing at a Geostationary Satellite. Figures taken from the TRW report and describing the referenced missions are included here for completeness. For all of these reference missions except number 3, the OMV (or OTV for mission 4) was used in a teleoperated mode. This in fact resulted in a high level of IVA for the Space Station astronauts supporting the servicing operations. The TRW report indicates that two IVA hours are used for each EVA hour in their referenced Satellite Servicing missions.

1. SCENARIO HIGHLIGHTS
   - OMV RETREIVES GRO FROM 400 KM ORBIT
   - RENDEZVOUS AND BERTHING AT SS
   - COMPREHENSIVE GRO STATUS TESTS
   - REPLACEMENT OF FAILED UNIT(S)
   - PROPELLANT REFILL
   - GRO CHECKOUT AND REDEPLOYMENT



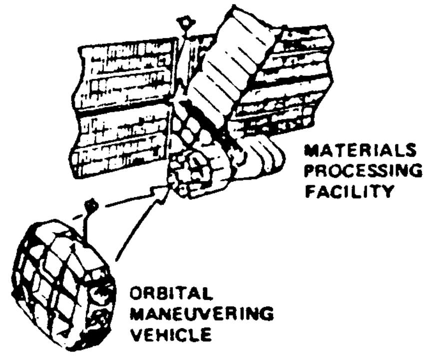
2. AUTOMATION REQUIREMENTS
   - REMOTE CONTROL OF GRO RETRIEVAL
   - AUTOMATED RENDEZVOUS AND DOCKING AT SS
   - LOAD HANDLING AND TRANSFER BY TELE-OPERATION
   - PROPELLANT REFILL
   - AUTOMATED TESTS, CHECKOUT, COUNTDOWN
   - DATA SYSTEM SUPPORT (DATA DISPLAY, DIAGNOSTICS, TROUBLE SHOOTING)

3. ACTIVITY COUNT
   - ESTIMATED ELAPSED TIME 10.5 HR
   - ESTIMATED TIME SAVING THROUGH AUTOMATION 10 HR

Figure 3. Reference Mission No. 1
Servicing GRO Satellite on Space Station

1. SCENARIO HIGHLIGHTS

- OMV ATTACHED TO SERVICING MODULE CARRYING FRESH SAMPLE MATERIAL
- OMV TRANSFERS TO AND PERFORMS RENDEZVOUS, BERTHING AT MPF
- SERVICER EXCHANGES SAMPLE MAGAZINES AT MPS UNDER REMOTE CONTROL
- OMV PERFORMS MPF ORBIT REBOOST
- RETURNS TO SS, DELIVERS FINISHED SAMPLES
- OMV REFURBISHED FOR NEXT USE



2. AUTOMATION REQUIREMENTS

- LOAD HANDLING AND TRANSFER AT SS BY TELEOPERATION
- RENDEZVOUS, DOCKING/BERTHING
- SAMPLE MAGAZINE CHANGEOUT
- MPF ORBIT REBOOST BY OMV
- AUTOMATED CHECKOUT, COUNTDOWN

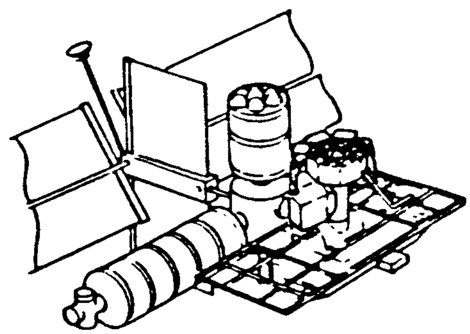
3. ACTIVITY COUNT

- ESTIMATED ELAPSED TIME 4.8 HR
- ESTIMATED TIME SAVING THROUGH AUTOMATION 7.0 HR

Figure 4. Reference Mission No. 2
Servicing Free-Flying
Materials Processing Facility (MPF)

1 SCENARIO HIGHLIGHTS

- INSPECT PAYLOAD/SUBSYSTEM TO BE SERVICED
- CALL FOR AND RECEIVE REQUIRED PARTS OR SUPPLIES VIA ORBITER
- TRANSFER SERVICING OBJECT TO AND FROM WORK STATION
- PERFORM REPAIR, REFURBISHMENT, MODULE REPLACEMENT
- CHECKOUT AND RESTORE TO NORMAL OPERATION



2. AUTOMATION REQUIREMENTS

- LOAD HANDLING AND TRANSFER
- AUTOMATED TESTS, DIAGNOSTICS, CHECKOUT
- MODULE REPLACEMENT BY TELEOPERATION

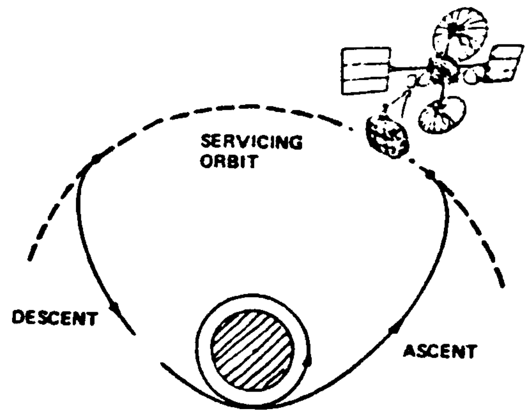
3. ACTIVITY COUNT

- ESTIMATED ELAPSED TIME 2.9 HR
- ESTIMATED TIME SAVING THROUGH AUTOMATION 3.9 HR

Figure 5. Reference Mission No. 3
Servicing of Space Station-
Attached Payload or Subsystem

1. SCENARIO HIGHLIGHTS

- CALL FOR AND RECEIVE NEEDED SUPPLIES VIA ORBITER
- ATTACH SERVICING MODULE TO OTV
- TRANSFER TO SYNCHRONOUS ORBIT, RENDEZVOUS AND DOCK WITH TARGET SATELLITE
- CHECKOUT, REPLACE FAILED MODULE AND/OR REFUEL SATELLITE
- RETURN TO SS (POSSIBLY BY AEROBRAKING MANEUVER)



2. AUTOMATION REQUIREMENTS

- LOAD HANDLING AND TRANSFER ON SS
- ASSEMBLE SERVICING VEHICLE WITH OTV
- AUTOMATED CHECKOUT, COUNTDOWN
- ORBIT TRANSFER, RENDEZVOUS, DOCKING/BERTHING
- INSPECTION
- MODULE REPLACEMENT
- REFUELING

3. ACTIVITY COUNT

- ESTIMATED ELAPSED TIME 11.1 TO 13.1 HR
- ESTIMATED TIME SAVING THROUGH AUTOMATION 6.1 HR

Figure 6. Reference Mission No. 4
Servicing Geostationary Satellite in Situ

Initial Space Station operations will probably not have benefit of an operational OMV and early OMV operations when the system becomes available will likely be limited in productivity while teleoperations experience is being acquired. For those reasons a period of time will probably exist when EVA is the primary method of performing satellite servicing. Because of that, we need to address issues of how to facilitate an EVA for Space Station which is easy for the astronauts to access and comfortable so that day in - day out operations are low overhead and physically acceptable from an operators point of view. The overhead costs in time and resources are only part of the penalties of don/doff constraints, prebreathing and suit restrictions. Those astronauts that will be going on EVA five or six days per week are rightfully going to want an operator friendly system. So one of the first answers to our evolutionary question is that the path starts with a more versatile and comfortable EVA system.

Indications on how the EVA system could be made more versatile and comfortable can be drawn from the routine work world of undersea operations. In that world the diver has begun to move away from the anthropometric suit and into vehicles such as the WASP and Deep Rover. In those vehicles the diver is encapsulated in the environment of the surface and is equipped with manipulators and translation/attitude systems which are controlled from inside the unit.

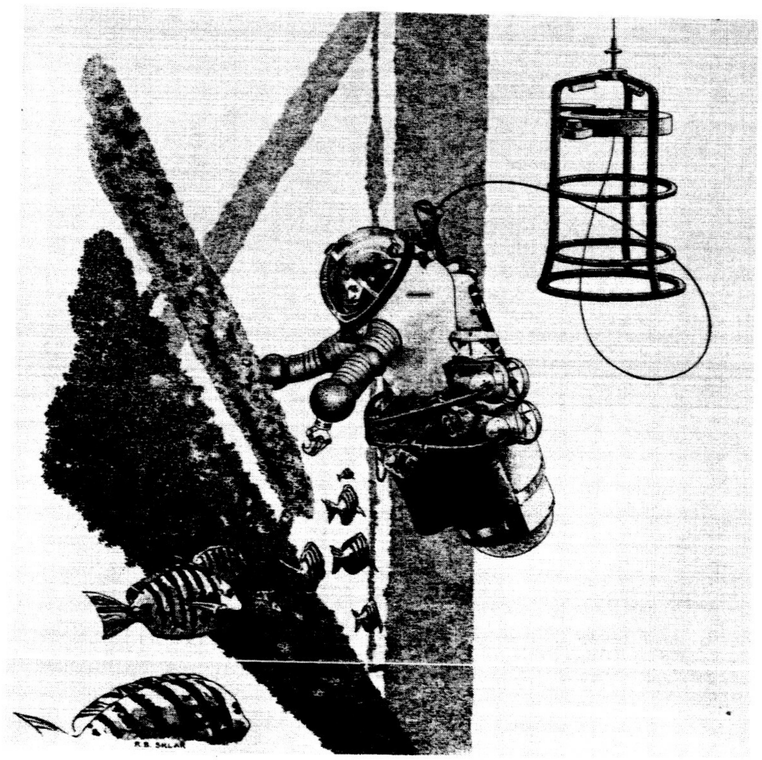


FIGURE 7

An EVA system which provides such a unit can have the unit dock to the Space Station at the neck ring and that would allow the astronaut to enter for EVA without donning a space suit. The pressure in the unit could be the same as the space station because leakage would be controlled by eliminating most anthropometric joints and their flexible sections. In short, EVA could be accomplished in an extension of the inside Space Station environment to the mobile work unit.

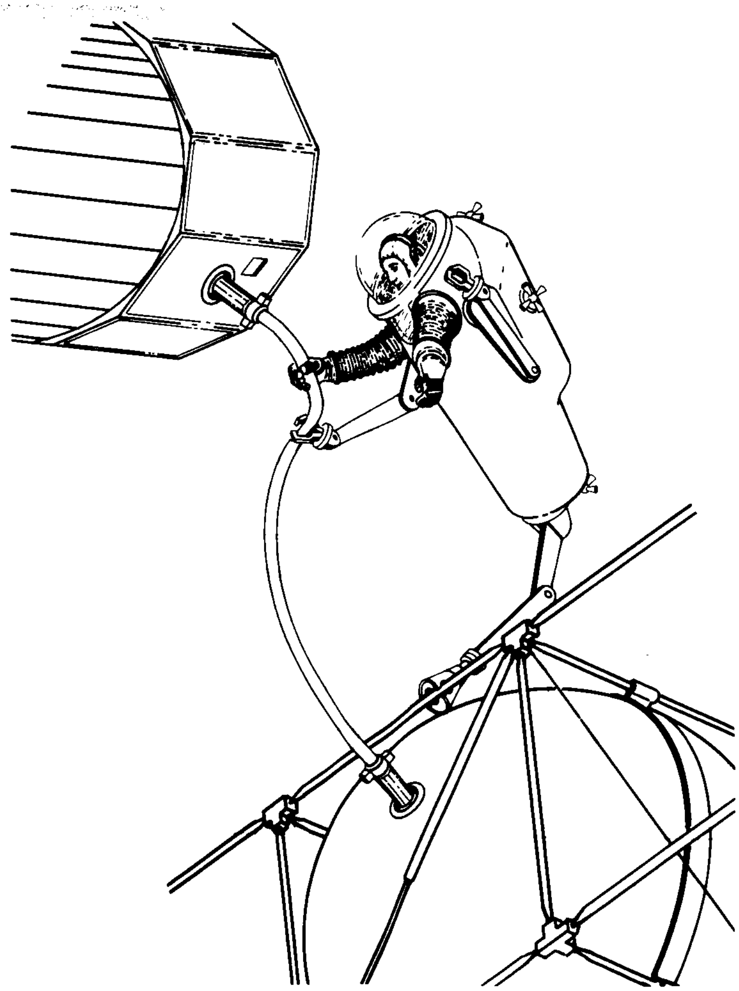


FIGURE 8

In pursuing our evolutionary question, our interest in such a mobile work unit is in the associated use of manipulators and position control at the task site. Such a unit could, as the Deep Rover does in undersea work, use coordinated manipulator control using a hand controller or joy stick. Such a system could integrate the control of the manipulator arm(s) with the grappling arm for control of attitude and work site positioning of the unit. The algorithms that would be used to translate hand controller signals to the multiple actuators of the manipulating arms/end effectors and the grappler arm would be precursors of the algorithms for remote commanding of a teleoperated robot. The on-site experience of EVA operators and their evaluation of the performance of the algorithms for control of mobile work unit manipulations and task orientation would aid the development of algorithms for robotic systems. The incorporation of machine decision making software to assist the encapsulated EVA operator by making decisions such as when the grappler arm moves versus when the manipulator arm adjusts would be a step toward autonomous robotics. In summary the use of an encapsulated work unit with manipulators and grappler control through a joy stick would not only make EVA more versatile but would enhance the development of robotics as well.

As indicated in our discussion of the TRW results a role will be played by teleoperated robots such as the OMV and OTV in support of satellite servicing. Establishment of such a capability will require the development of vision and tactile feedback technology as well as the manipulator and maneuvering control algorithms. As indicated above the algorithms for a teleoperated robot could evolve from a joy stick operations with a mobile EVA work unit. Currently a system which employs aspects of the commanding capability for teleoperators is the shuttle RMS. Indeed one of the most popular areas for initial Space Station automation and robotics advancement is in improving the RMS with more dexterous manipulators, mobility and telepresence feedback to the operator. The stereoptic vision and tactile feedback technologies needed for telepresence (or transmittal of work site sensations to a remote operator) are the subjects of rather intense technology advancement activity at this time.

As the technology evolves for the teleoperated robotic approach, the result will be an expanded capability for remote manipulators. The operators will have a better feel for the environment and configuration of the remote worksite and better control of the manipulations. Presumably new manipulation capabilities will evolve as the robot mechanisms advance and they will facilitate a larger task menu. The crew members using the system become more productive through those advancements, but the teleoperated system will not release operators for other work. The evolutionary path to more advanced teleoperations does not naturally advance technologies needed for autonomy of the robot from the operator (reference 5). It is likely that the time of operator involvement with the teleoperated system will actually increase as the system evolves because of a larger menu of tasks which the system is capable of performing. Crew performance with the evolving teleoperated system improves for the particular tasks performed with the system but the overall productivity of the space borne humans is not clearly advanced by such evolution. The problem is to free the humans from the machines so that the humans can perform the supervisory and goal setting roles which support expansion of Space Station operations and missions.

While teleoperations will probably not become obsolete because there will always be needs for intricate manipulations at hazardous sites which must be conducted remotely, the goal of increasing human productivity is clearly served by using autonomous robots where possible.

At some point then in the evolution toward robotics, there would have to be a radical change to an autonomy-based system if technologies were exclusively developed through teleoperations up to that point.

As autonomous robots become operational they could be used to assist EVA astronauts in a number of ways. Initially a voice controlled flying eye robot could be used to perform routine inspections outside of the Space Station and transmit images to an astronaut inside. The technologies to accomplish such a function are within reach but confidence in such a system operating near a spacecraft needs to be developed. Boeing is currently working on a simulation for such a flying eye robot for early demonstration of the concept. The next figure is a sketch of an early concept for such a flying eye robot demonstrator. In the concept we are working on, the robot would be directed by "forward", "back", "right", "left," "up", "down" commands to "fly" over the surfaces of the parent space vehicle. Such directions would be given by an astronaut who could keep the robot in view as it was directed on its initial tour. Fixes on planted navigational markings such as bar code strips would be taken at frequent intervals during the guided tour. The software program for the system would organize the tour fixes into a data base representing the "world" that the robot is to move through. After the "world" data is established the astronaut would be able to direct the robot by specifying a destination in terms of a navigational marking end point or a series of points along a path. The camera carried on the robot could be controlled by pointing and zoom commands from the astronaut.

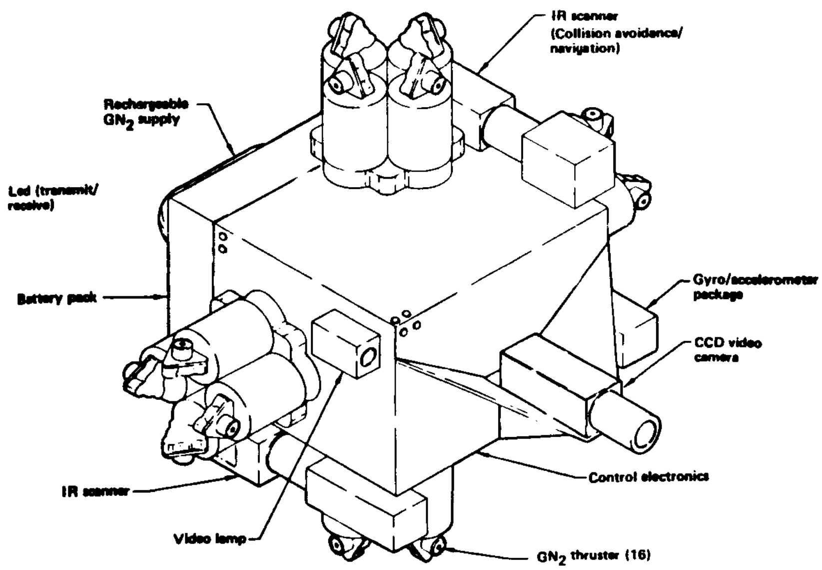


FIGURE 9

As the autonomous robot concept matures the voice directed robot could be used as an assistant for EVA astronauts performing satellite servicing tasks. In this use the robot would position itself according to voice commands from the EVA astronaut and perform tasks such as directing a light or providing a mobile caddy-like container for tools that the astronaut could use. In a more advanced mode the caddy could be directed to return to an external tool shed to "pick up" tools or parts and bring them back to the task site or to take position at some remote viewing site and transmit images to a display at the work site. In still more advanced operations a mature robot could hold and manipulate task elements under direction from the on-site EVA astronaut. The eventual goal for autonomous robots would be to conduct simple repetitive, assistance or hazardous tasks independently so that for astronauts time on EVA could be reduced.

This paper then concludes with the position that evolution toward robotics must start with a more versatile EVA but lead to autonomous helper robots supporting EVA astronauts in servicing satellites.

Summary of Conclusions

The conclusions of this paper are summarized here as follows:

1) Space Station satellite servicing missions will be heavy time users and productivity motivates use of robotics.
2) Until the OMV and autonomous robots become operational, human EVA will be needed to perform satellite servicing missions.
3) Use of mobile work capsules would provide versatile human EVA and joystick control of manipulators leads to robotics.
4) Autonomous robots could evolve to perform useful tasks in support of satellite Servicing.
5) Evolutionary paths to robotics should proceed through autonomous robot development as well as through teleoperations.
6) The goal of increasing human productivity is served by using autonomous robots for tasks that can be performed by them.

References

1) Aviation Week and Space Technology - March 26, 1984, pages 42-51.

2) Aviation Week and Space Technology - Sept. 9, 1985, pages 21-23.

3) Boeing's Advanced EVA System Design Requirements Study Report to be published Dec. 1985.

4) TRW Z410.1-84-160 "Space Station Automation Study; satellite servicing" Nov. 30, 1984.

5) AIAA 22nd Space Congress Paper "Evolutionary Paths for Artificial Intelligence Technologies in Operator Interactions with Space Robots" given by Ron Hammond of BCS, April 25, 1985.

SERVICING WITH SMART END EFFECTOR ON OMV MANIPULATOR

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ABSTRACT

This paper describes a smart mechanical hand developed at the Jet Propulsion Laboratory (JPL) for experimental use and evaluation on the Prototype Flight Manipulator Arm (PFMA) at the Marshall Space Flight Center (MSFC). PFMA-type arms have been considered in the past to be part of the Orbital Maneuvering Vehicle (OMV) which will be employed for satellite servicing and other Space Station related operations. The paper first presents general design and performance criteria that will lead to enhanced multifunctional operation capabilities. This is followed by the summary of specific design requirements used in constructing the OMV hand and the mechanical design description. Space robot hands require the use of multiple sensors integrated into the mechanical hand for task supervision and control. Sensing, electronics, the distributed microcomputer control and the operator control interface and graphics displays are then discussed.

TABLE OF CONTENTS

I. INTRODUCTION

Anticipated construction, assembly, servicing and repair tasks in Earth orbit will require the use of multifunctional remotely controlled manipulators. Multifunctionality in remote manipulation to a large extent resides in the mechanical, sensing and control capabilities of the mechanical hand integrated both mechanically and in the control with the manipulator, including the man-machine interface. Using the human analogy, one should recall that the hand is a powerful and delicate tool as well as a tool manipulator, but also a sensory organ through which information is received and transmitted. Furthermore, the function of the arm is to position and orient the hand, act as a mechanical connection and as a power and sensation transmission link between the hand and the main body of a person. The full meaning of the arm is revealed by the hand.

Motivated by the considerations above, a brief overview of general design requirements is presented in Section 2. The general design requirements summary looks at multifunctional hands as integrated subsystems and, therefore, covers their mechanical, sensing and control aspects.

Specific design and performance requirements for the OMV smart hand are summarized in Section 3. The requirements are based on anticipated characteristic tasks to be performed by the mechanical hand and on considerations for advancing the state-of-the-art in sensing-based hand control.

The overall smart hand system including control and operator feedback is introduced in Section 4. Sections 5 through 9 are detail descriptions and illustrations of the individual subsystems. They are:

- The mechanical design details including the intermeshing claw configuration that was conceived to handle the perceived tasks as well as to accommodate local sensors and electronics (Section 5).

- Several critical drivers influence the design, engineering and integration of these sensor/computer elements. These criteria include high data rate from the multiple-channel sensors, transmission of power and data over limited physical paths, compactness of design, minimization of power/volume, and shared manual/automatic control of the robot hand. The design and control of such a multi-sensor robot hand poses quite a challenge. Four different types of sensing and control are designed into the hand: (a) force and torque sensing at the wrist, (b) clamping force sensing at the fingers, (c) proximity sensing within and outward from the claw, and (d) position and rate sensors (Section 6).

- Multiple microprocessors are designed into the local electronics, local at the robot hand, for the inner loop control, sequencing, and data acquisition. In turn, these microprocessors communicate and receive control commands from the central computer system, which also consists of multiple microprocessors (Section 7).

- o The control and operator control interfaces are described in Sections 8 and 9.

II. GENERAL DESIGN REQUIREMENTS

The general hand design requirements can be subdivided into four principal areas: (i) mechanical design and performance, (ii) sensing, data acquisition and transmission, (iii) control, and (iv) man-machine interface for decision and control.

A) Mechanical Design and Performance

The clamping system determines the gripper configuration. Major general purpose end effector categories are shown in Figure 1. Increasing end effector multifunctionality requires progress in the direction of the arrows. The OMV smart hand is an effort to increase end effector sophistication through enhanced sensing. But as can be seen in Figure 1, dexterous manipulations require technology progress at two levels, the other being an increase in the degrees of freedom of the end effector to enhance capabilities from a mere clamping of objects to object manipulations. Thus, our next hand development will be a multi degree of freedom hand (Reference 1).

Manipulative capabilities are defined as the skill to manipulate objects with a multi degree of freedom hand, i.e., turning an object in the hand or squeezing a trigger. It requires either redundancy in clamping possibilities or a 2 degree of freedom knuckle joint so that the finger can move or clamp in different directions.

Dexterity is defined as the combination of manipulative capabilities and smartness. The anthropomorphic hand is a dexterous hand in human hand shape. Since autonomous dexterous hand manipulations are still at least a decade away, the anthropomorphic hand is an important near term solution: a sensed human hand can be used as input device to manipulate the mechanical hand in an efficient and user friendly way.

1. Hand Design Alternatives

a. One degree of freedom end effectors

i) The base model

The current state-of-the-art of general purpose EE designs is the two finger gripper with a linear closing motion. Its primary advantages are the simplicity in design and the available technology. However, it has very limited multifunctionality with clamping capabilties only, no manipulative capabilities, no dexterity and it can handle only a narrow range of object sizes and shapes. Tool handling is very restricted, too, to a mere holding of tools. Proper object and tool presentation and orientation for grappling is necessary.

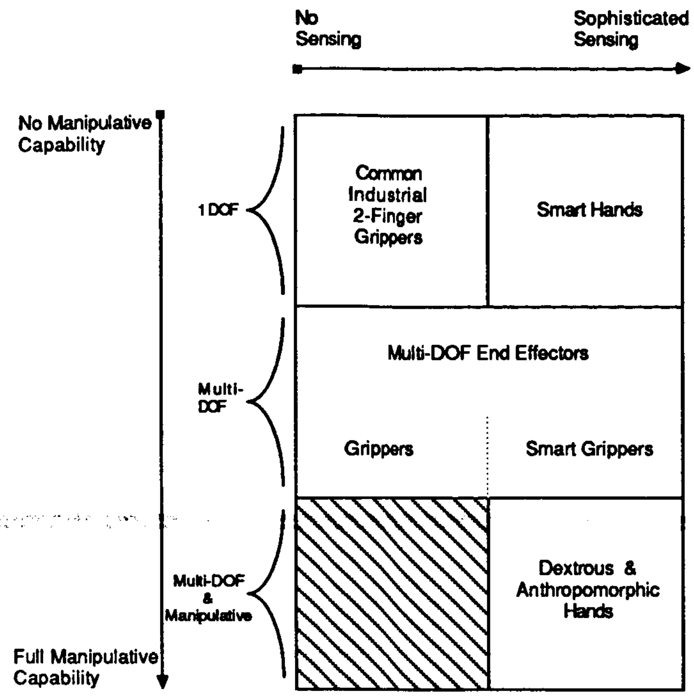


Figure 1. End Effector Categories

ii) Smart Hands

Sensor integration into this hand will aid in object recognition and grasping. The hand is thus referred to as a smart hand. But sensors increase the size of the hand so that it becomes very bulky.

iii) Exchangeable Smart Hands

In order to increase the systems capabilities it is often suggested to use exchangeable special purpose smart hands of different sizes or functions. Even though a larger range of objects could thus be handled, there still is no dexterity for skillful object manipulations while costs become prohibitive for developing and space qualifying a number of exchangeable end effectors together with creating reliable exchange interfaces and end effector racks with docking/undocking in proper orientation and tethering mechanisms at the robot arm and at each stowage location.

Any full end effector exchange would require the following complex links between the exchangeable end effector and the permanent base:

- Data lines for commands to the end effector
- Sensor data lines (vision, force, torque, etc.)
- Electrical power or mechanical drives
- Structural connections

- Coupling, aligning and centering mechanisms
- Thermal blanket handling capability
- Automatic shutters or lids at the exchange interfaces to protect the mechanism.

Other considerations are weights and stowage space, reliability of the individual exchange mechanisms, the number of exchanges needed per task and the time it takes to interrupt the operation and move the arm to the stowage bin to get another end effector. The above factors will make automatic gripper exchanges a rather unlikely task, if not totally infeasible.

Recent suggestions for space station robot end effectors seem to lean toward the construction of exchangeable end effectors because nobody expects a real breakthrough in deterous hand design within the next 5 to 10 years. But the experts agree that if dexterous hands were available, they would be so much more capable for any kind of advanced robotics. More capable (dexterous) end effectors are needed in the near future for applications which normally require EVA performance. JPL is planning to undertake an effort to construct dexterous hands.

Considering the development costs of one anthropomorphic hand vs. several simpler smart hands plus exchange mechanisms, end effector stowage facilities with their mechanisms and the need for proper object orientation to be able to latch on to the arm interface, it does not need much evalation to realize that one sophisticated compact design will be much more economical in the long run.

b. Dexterous Hands

Looking at human hands as models for dexterous hand constructions shows the mechanical characteristics that hand designers ought to adapt to combine several end effectors into one compact design: i) It has fingernails which can be used for scratching and probing or dispensing adhesive tape to attach heat blankets. Special fingertip inserts could also be used. ii) Individual fingers serve for precision work and for manipulating small objects where the palm can lean against the object structure for support so that very accurate fine-adjust manipulations can be performed. iii) Several fingers acting together increase load capability and, with their flexibility, objects can be hugged. They also should be able to handle Velcro straps and heat blankets. iv) The palm is used for heavy loads where the load is applied near the wrist and may rest at the arm to reduce moments and to increase stability. In this mode, the fingers are used mainly to clamp the object rather than lift it.

There are three key issues to be resolved for the successful implementation of a multi degree of freedom end effector (Reference 1):

i) The complex mechanical design.

ii) The capability to control the many degrees of freedom to execute coordinated hand motions.

iii) Active Mechanical Compliance

It is the human muscle equivalent capability to tighten or loosen a muscle which acts as the joint stiffness control. In the "soft" mode, a compliant limb will yield to outside forces; in the "stiff" mode, the limb will resist yielding. As a practical consequence, a hand commmanded to close over an object will conform to the object's shape. The hand can then be stiffened, and clamping force applied, enabling a much better grip on the object.

A dexterous hand/arm system can plug its own arm cables into the sockets for power supply and communication links at the interface for applications such as using the dexterous hands on the shuttle's RMS arm. Then it will unplug the similar connectors at the stowage location or vice versa. Thus, the system is always powered up and can establish its own docking without the need for any automatic coupling features.

2. Tool Usage

It is obvious that no single hand design can accommodate all requirements to successfully handle all objects. Even the most sophisticated end effector, the human hand, uses a variety of manual and power tools and still needs other aiding devices for even quite common tasks.

Employing tools was the turning point that changed early man's life. It will have the same effect on robot hands where the usage of tools will enhance the robotic capabilities and application ranges. It is therefore important to realize that the successful manipulation of tools by the robot hand is one of the most important design criteria.

One hand alone cannot accomplish much by itself. Therefore, the final configuration of smart robot hand systems will be a multi-handed configuration where the hands assist each other.

Two different sets of requirements exist for end effectors, depending on the dexterity level of the hand:

a) Non-Dexterous Hands

The criteria for non-dexterous hands are somewhat similar to those for exchangeable end effectors. Development of a specially designed tool stowage facility is needed where each tool must be presented in the proper orientation for grasping. The stowage facility will be bulky and needs automatic tool docking, locking and tethering mechanisms at each stowage location.

b) Dexterous Hands

Characteristics of tool manipulations with dexterous hands:

o Off-the-shelf tools can be used with minor modifications

- o The currently used EVA tool stowage can be used. without modifications since the tools can be grasped in any orientation (for instance from Velcro strap surfaces or in styrofoam holders) and manipulated until they are rigidly aligned in the hand.
- o A second dexterous hand can assist in grasping and holding of tools.
- o The tether can be attached to the security ring with the other hand. No automatic tether coupling mechanism needs to be developed.
- o The hand/arm has built-in compliance required for many tool manipulations.
- o Powered tools can be plugged into a socket on the tool holding arm with the other hand.
- o Triggers can be squeezed with one finger.

B) Sensors

Intelligent operations require a great amount of sensory information which includes force, moment, position, tactile, temperature and proximity sensing, object recognition, global and local vision and many more. Space permitting, any number of sensors can be built into the hand. Much work is needed to downscale the sizes of sensors, for most of them are far too bulky for practical applications within or at the hand.

If possible, sensors and feedback routing should be placed entirely within the physical confinements of the hand for protection. Otherwise, contaminents and moisture inflow might hamper their operations or material handling may crush them if located in exposed positions. Tactile and any other sensors which are located on the surface need to be sealed and extremely rugged. The amount of sensory feedback will determine if local preprocessors are needed. Multiplexing will be necessary with advanced hands.

C) Control

Robots do not yet have the capability to adjust to major changing situations. A human operator is therefore required in the control loop to make all major control decisions. Artificial intelligence will eventually help but is still years away in its development. With human operators controlling the teleoperation system, the control station must present the pre-evaluated feedback to the operator in easy-to-understand form for quick recognition, comprehension and decision-making by the operator.

D) Man-Machine Interface

The information flow between the operator and the teleoperator system is a presentation of sensed information to the operator and the operator's control decisions back to the controller.

With vision being the most important sense, a visual signal in the form of a mono or stereo TV picture will have to be transmitted to the operator from the robot. It will provide the operator with a sense for where the hand is reaching. Additional cameras might be mounted at the arms or in the hand of the robot to aid in grasping. Other sensory information can be presented in graphic, acoustic or some other form that provides convenient state evaluation possibilities for the operator.

Mechanical, electromechanical and electrical interfaces are common in master-slave arrangements. Positional control will be simplified if the operator manually performs the motion which the end effector will repeat. This positional control can be done in a master-slave control arrangement. The master-slave arrangements should incorporate as many feedbacks as possible right to the hand-arm system as possible (i.e. force or position reflecting, tactile feedback) so that the operator's visual attention can be directed fully to monitoring the optical feedback from the TV system.

III. SPECIFIC DESIGN AND PERFORMANCE REQUIREMENTS FOR THE OMV HAND

The specific design and performance requirements for an OMV/PFMA smart hand are derived from (i) considerations of typical tasks the hand has to perform, (ii) considerations of the system the hand has to be interfaced with, and (iii) considerations of advanced sensing, control and man-machine interface capabilities which should be demonstrated and tested for performance evaluation. The task and system interface requirements were essentially provided by MSFC. The advanced sensing, control and man-machine interface requirements are essentially results of base technology development at JPL.

A. Test Tasks

Typical test tasks are represented by the following categories which are related to construction and repair in Earth orbit:

1. Mate and demate a fluid coupling mechanism which has an open area of 10.2 cm by 8.6 cm (~4 inches by 3 and 3/8 inches) to reach the handle.

2. Open and close an access panel by turning a wing nut which is a 0.5 cm (~3/16 inches) thick flat stock with an area of 7.6 cm by 3.8 cm (~3 inches by 1 and 1/2 inches).

3. Remove and replace a battery module by grasping a square beam handle.

4. Deploy and retrieve a telescoping vertical antenna.

These test tasks dictate the following requirements:

a) The hand shall have an outside width no wider than 18 cm (~7 inches).
b) The maximum hand opening shall be not less than 6.4 cm (~2 and 1/2 inches).

c) The minimum hand closing shall be no more than 0.6 cm (~1/4 inches).
d) The overall construction of the hand shall be so that the hand can reach into the fluid coupling mechanism.
e) The hand shall be capable of squeezing with 445 N (~100 lb) force.
f) The maximum tip force on the hand shall be at least 45 N (~10 lb) and the maximum tip torque shall be at least 20.3 Nm (~15 ft-lb).
g) The maximum closing velocity of the hand shall be at least 2.5 cm/sec (~1 inch/sec).
h) The gripping action shall have a linear path throughout the travel to prevent preloading of an object as it is being grasped.

B. System Interface

1) The entire mechanical hand system shall mount to the PFMA wrist in accordance with detailed drawings of Martin Marietta Co.

2) The general size of the mechanical hand and of the claws shall be similar to detailed drawings of Essex Co. In particular, the claws shall be intermeshing such that oval, round and square beams as small as 0.6 cm (~1/4 inches) in diameter can be grasped.

3) All electrical communication to and from the mechanical hand, including the electrical power, shall be through an existing slip ring subsystem at the last wrist joint. This slip ring subsystem permits the use of altogether seven electrical wires for power and/or signal transmission.

4) The whole control and display data handling system of the smart hand shall be interfaceable to the control computer and display system of the designated MSFC control station both functionally and operationally.

C) Advanced Sensing, Control and Man-Machine Interface

1) The force-torque sensor mounted to the base of the mechanical hand shall measure forces and torques as applied to the hand in all three orthogonal directions up to 133 N(~30 lb) and 68 Nm (~50 ft-lb) in each direction with a resolution of at least 1 part in 500.

2) The grasp force sensor mounted to the base of the claws shall measure grip force up to 535 N (~120 lb) with a resolution of at least 1 part in 200.

3) Electro-optical proximity sensors mounted to the claws shall measure short distances up to at least 6.3 cm (~2 and 1/2 inches) inward and outward relative to the surface of the claws. The number and geometric arrangement of sensors shall permit measurement of as many as possible task degrees-of-freedom (3 position and 3 orientation coordinates) of the hand relative to objects and environment. The distance resolution of individual sensors shall be at least 1 part in 50 (about 1 mm or 5/100

inches).

4) The grasp control loop shall be closed locally at the mechanical hand based on commands from the central control computer at the MSFC control station.

5) The computer graphics display of sensor data shall permit (i) the use of alternative display formats on the task level and (ii) the fuse of computer graphics with video data on TV monitors. The computer graphics update rate shall be at video update rate on a color display.

6) The routing of sensor data shall permit the use of sensor-referenced control through the MSFC control station computer.

7) The smart hand drive motor shall be a three-phase brushless DC torque motor to minimize electrical noise to the nearby sensor and drive electronics.

IV. OVERALL SMART HAND SYSTEM

Overall smart hand system major functional blocks are shown in Figure 2. The JPL-developed smart hand system elements are within the two areas defined with dotted lines in Figure 2.

The basic operation mode requires that the operator give system commands at the control station and observe response parameters on the graphics display. The control computer formats these commands to a serial data stream that is transmitted to the end effector controller via one of the seven slip rings.

Power for the end effector is supplied by a power supply located at the base of the PFMA and transmitted via four of the seven slip rings. System ground is connected via another slip ring. Serial data from sensors on the end effector is sent over the seventh slip ring to the signal processing computer also located at the base of the PFMA. This data consists of readings from the force torque sensors, the proximity sensors, and position, velocity, and force of the gripper. The signal processor computes the parameters of interest from the raw data and feeds this information to the graphics processor. The graphics processor then generates the video signal which will display the parameters on the MSFC graphics display video monitor.

The end effector controller receives the commands from the control computer, interpreting the code and calculating the required response of the motor which drives the gripper. Commutation position information from the brushless DC motor is also an input to the end effector controller, which determines the proper drive signals and actuates the motor through the high current drivers. The motor is connected to the jaws of the gripper by a simple mechanical transmission element. The force exerted by

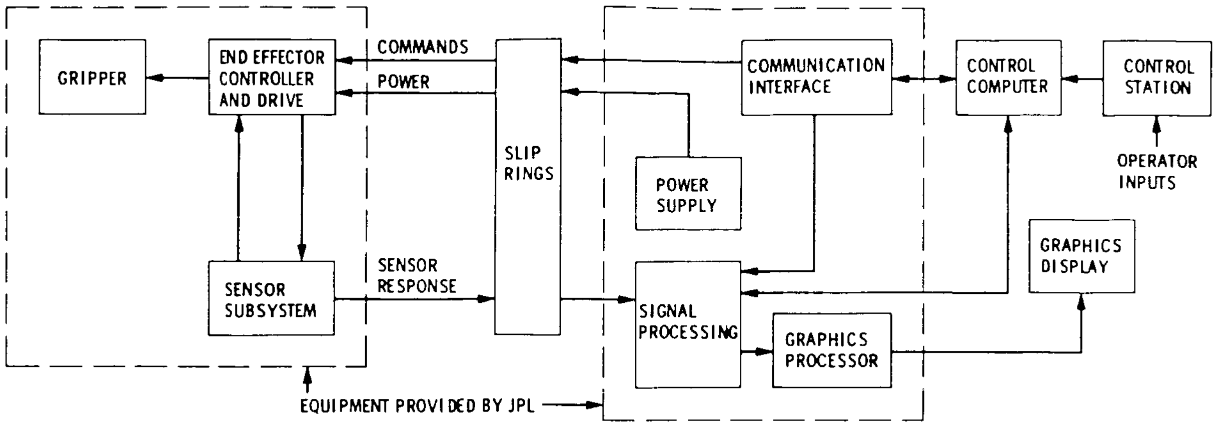


Figure 2. Overall Smart Hand System

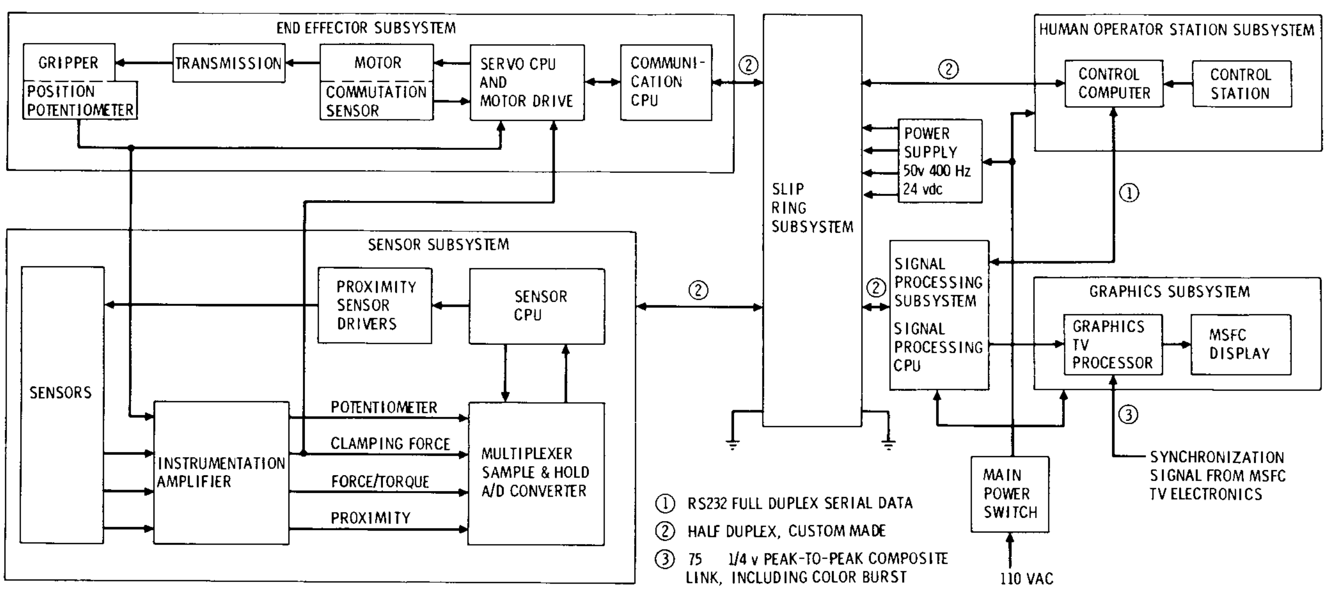


Figure 3. Smart Hand Overall Data Handling

the gripper is monitored by strain gages. This force information may be used for display purposes, for servoing the force to a commanded level and to limit the motor drive to not exceed a maximum force level.

The function of smart hand system components together with the overall data handling are shown in greater detail in Figure 3. Categorized by their distributed functionality, five subsystems are called out: (a) Sensor Subsystem, (b) End Effector Subsystem, (c) Signal Processing Subsystem, (d) Graphics Subsystem, and (e) Human Operator Control Station Subsystem. Each subsystem has its own microcomputer(s), thus forming a distributed microprocessor system. The distribution of the computing and processing power is necessitated by the large amount of data processing, and is dictated by the physical separation between the "local" (local to the hand) and "central" (central to human operator) electronics. As seen in this figure, the MSFC control computer receives command inputs from the operator at the control station and formats the commands into a serial stream of ASCII codes with a conventional RS-232 interface. Through this interface the commands are entered in under control of the servo CPU. The servo CPU compares the state of the end effector as indicated by the inputs of position, velocity, and force level from the sensor subsystem. If there is any difference, the servo CPU computes the appropriate motor drive signal to bring the system to its commanded state. By monitoring the motor rotation via the commutation sensors, the proper motor winding is selected and a pulse width modulated (PWM) signal is sent. The motor drive amplifier converts the PWM signal from the CPU into a high current drive signal to the motor winding. The motor torque goes through a simple transmission with a lead screw converting rotation to linear motion of the gripper jaws.

The sensor subsystem is controlled by the sensor CPU and provides other information on the end effector besides the position, velocity, and force feedback to the motor CPU. All of the information from the various sensors is selected by an analog multiplexer feeding a sample and hold input to an analog to digital converter (ADC). The sensor CPU issues the select signals to the multiplexer, initiates the hold command and conversion start for ADC operation, and receives the 12 bit data via a parallel input port.

Each jaw is equipped with a strain gage bridge to measure grasp force. Furthermore, there are eight strain gage bridges in the force-torque sensor at the base of the hand. Their outputs are amplified by instrumentation amplifiers before going to the multiplexer. The grasp force strain gages are similarly amplified. The proximity sensor operation is also controlled by the sensor CPU but requires additional sequencing and synchronization besides simply selecting the multiplexer input. The proximity sensor operates on the principle that light reflected from the workpiece surface varies with the distance of the surface from a light emitter-light detector pair. The sensor CPU operates an array of emitter-detector pairs by sequentially driving the emitter and simultaneously reading the corresponding detector via the multiplexer-ADC data path. All of the data gathered by the sensor subsystem CPU is formatted into a serial data stream and sent via one of the slip rings to the signal processing subsystem at the base of the PFMA.

The signal processing subsystem communicates with the sensor subsystem via a half duplex serial link. The signal processing subsystem converts the raw data from the sensor subsystem into parameters of interest for the system operator. These parameter values are formatted into display types, then sent as commands to the graphics processor for generating the displays.

The graphics processor has the option of receiving GENLOCK synchronization from the MSFC video system. This enables the graphics displays to be video compatible with standard TV monitors so that graphics data can be mixed with video data on the same monitor.

The video signal synthesized by the graphic processor inputs directly to the MSFC video monitor to produce the graphics displays. One of the display options is the force torque sensor measurements, which displays forces as distance along a three axis coordinate system shown in a perspective view. Another display option will show the proximity of the gripper jaws from the workpiece, or the grasp force when the gripper engages the workpiece. The monitor presents the selected displays to the system operator as part of the control station environment.

V. MECHANICAL DESIGN

The overall view of the smart hand mechanical assembly is shown in Figure 4. The main feature of this mechanism is that it has two support columns within a channel type support frame. Each claw is supported by two structural elements: a column and a sliding frame in the channel. Each claw has its own support column independent from the support column of the other claw. The drive mechanism is shown in some detail in Figure 5.

The mechanical design features can be summarized as follows:

- o Overall size considerably less than maximum permitted for MSFC design. Grip range is 8.8 cm (~3 and 1/2 inches).
- o Short length minimizes distance between wrist pitch/yaw axes and load for greater manipulator work envelope and load capacity.
- o Channel type frame produces a stiff structure.
- o One piece aluminum channel frame with integral bevel gear box produces a rigid structure.
- o Hardened steel bevel gear drive between motor (not shown) and ball screws produces a compact and efficient drive train.
- o Left hand and right hand ball screw mechanism drives double finger slides in a coordinated fashion.
- o Double slides, each slide on a separate hardened and ground steel rod, are additionally guided by in a channel integral with the frame.
- o Double slide rods supported at both ends produce a compact design.

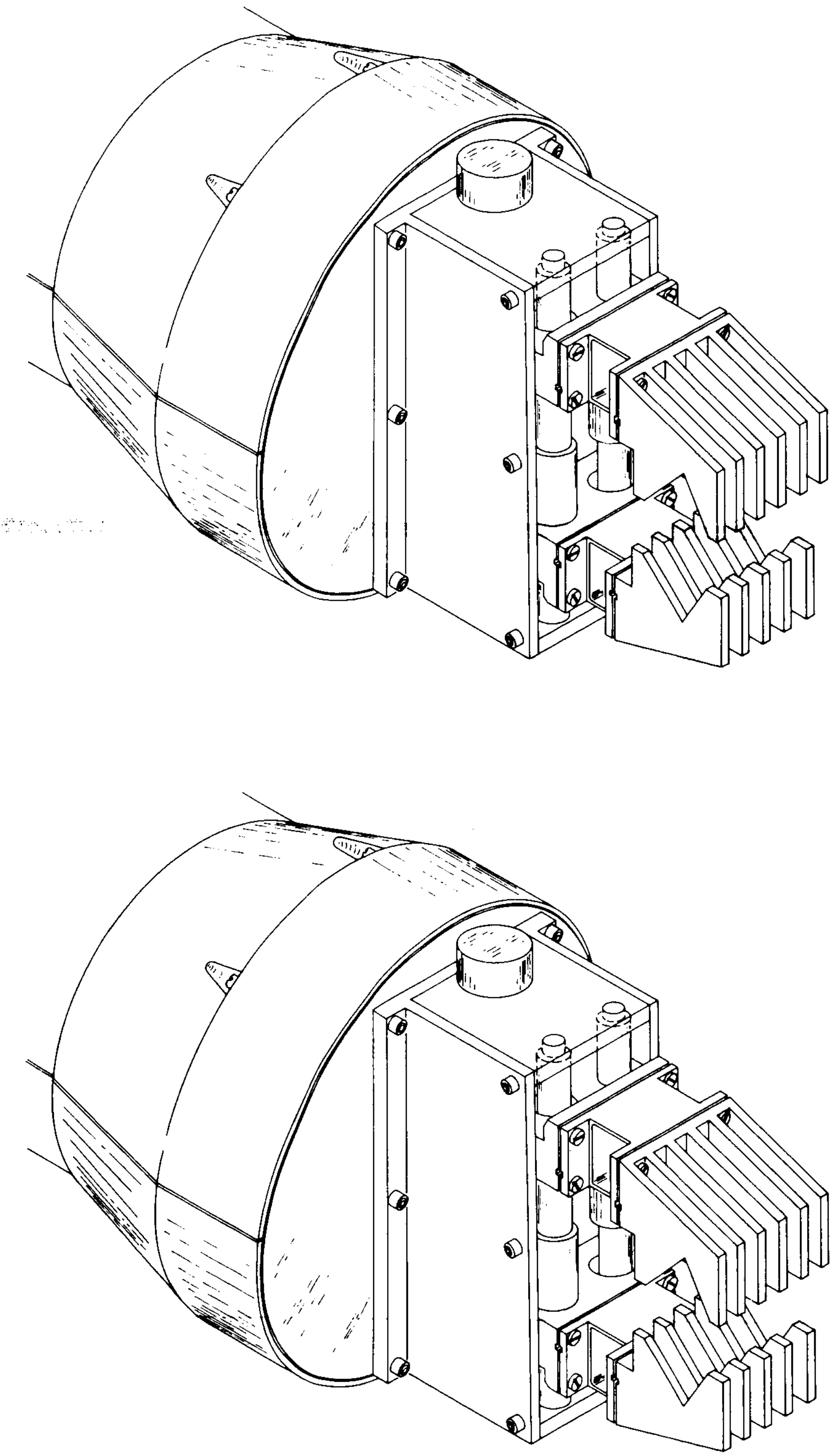


Figure 4. Overall View of Smart Hand Assembly

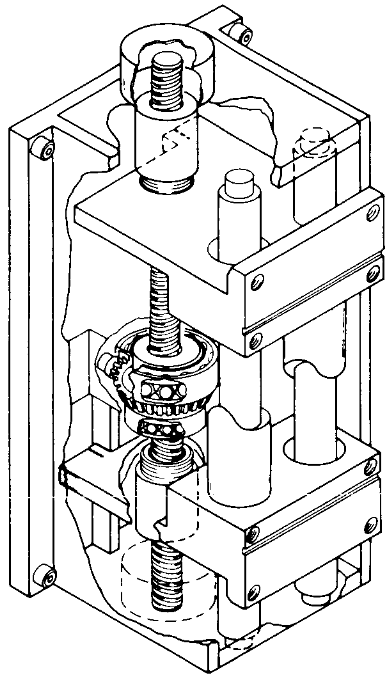


Figure 5. Smart Hand Drive Mechanism

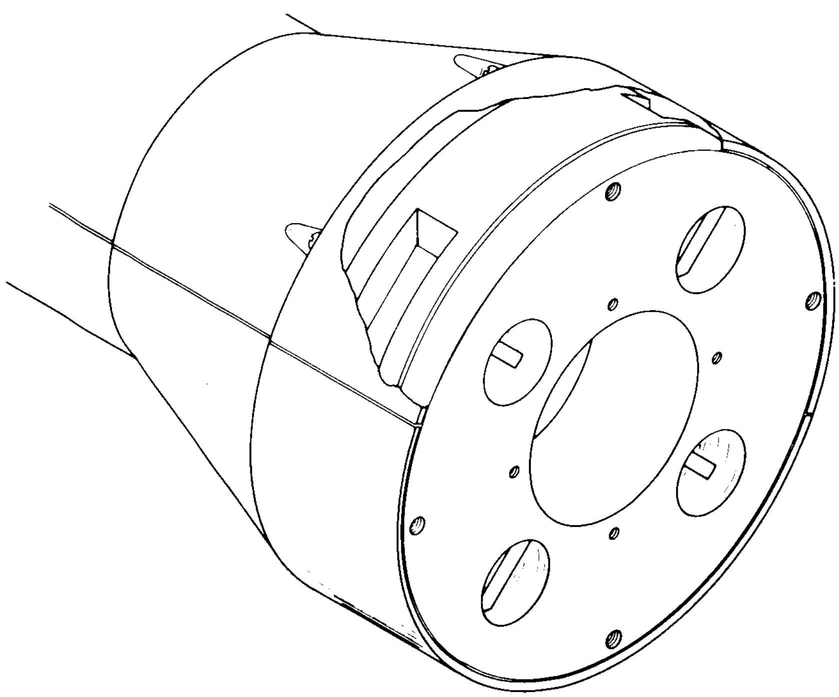


Figure 6. Smart Hand Wrist Force-Torque Sensor and Electronics Bay

- o Determinate design and built in adjustment features of slides minimizes tendency to bind and produces a gripper mechanism built of interchangeable parts with few ultraprecision dimensions.
- o Partially enclosed drive and slide mechanisms are resistant to damage.
- o High efficiency drive will not lock up under any conditions.
- o Bevel gear and ball screw drive efficiently matches motor to load.
- o Brushless Rare Earth Magnet D.C. motor is used for long life and compact power source.
- o Low power consumption reduces motor heating and simplifies drive electronics.
- o Motor can maintain maximum grip force continuously without overheating. Maximum grip force is 540 N (~120 lb).
- o Fail-safe brake on motor maintains grip in the event of power loss.
- o Gripper mechanism attaches with easily accessible screws.
- o Claw assembly and grasp force sensor easily changed.
- o Mechanism materials and bearing design may be changed for space rating the gripper.

The force-torque sensor mounted to the base of the end effector is shown in Figure 6 together with the electronics bay. The main features of this sensor design are summarized below:

- o Wrist force sensor resolves all six components of the resultant of forces and moments on the gripper. [Fx, Fy, Fz, Mx, My, Mz]
- o Sensor uses a Maltese cross type of design. All sensing is done by strain gage bridges on <u>bending</u> beams. This helps temperature compensation and design has very low tendency to buckle.
- o Large bore in center permits gripper motor to extend through sensor to produce a compact sensor/gripper package.
- o Flat washer type design keeps wrist/gripper length short for improved work envelope and load capacity.
- o Sensing and overload structure is machined from a single piece of high strength aluminum alloy. This produces a low hysteresis, adjustment-free sensing system with good stiffness. There are very few precision machining tolerances.
- o Overload protection is built in to withstand unexpected overloads and accidents.
- o Six inch diameter of the sensor matches it well to gripper mechanism.
- o This design has been used before and existing software can be easily adapted to this specific application.

The intermeshing claws with the grip force sensor are shown in Figure 7. The main features are as follows:

- o Sensor senses grip force only on each finger. The sensor is not sensitive to placement of load in claw or other forces or moments applied to or by the claw.
- o Parallelogram design produces nearly pure translation type of deflection rather than angular bending typical of many other simple cantilevered beam sensors. Jaws of claws remain parallel or nearly

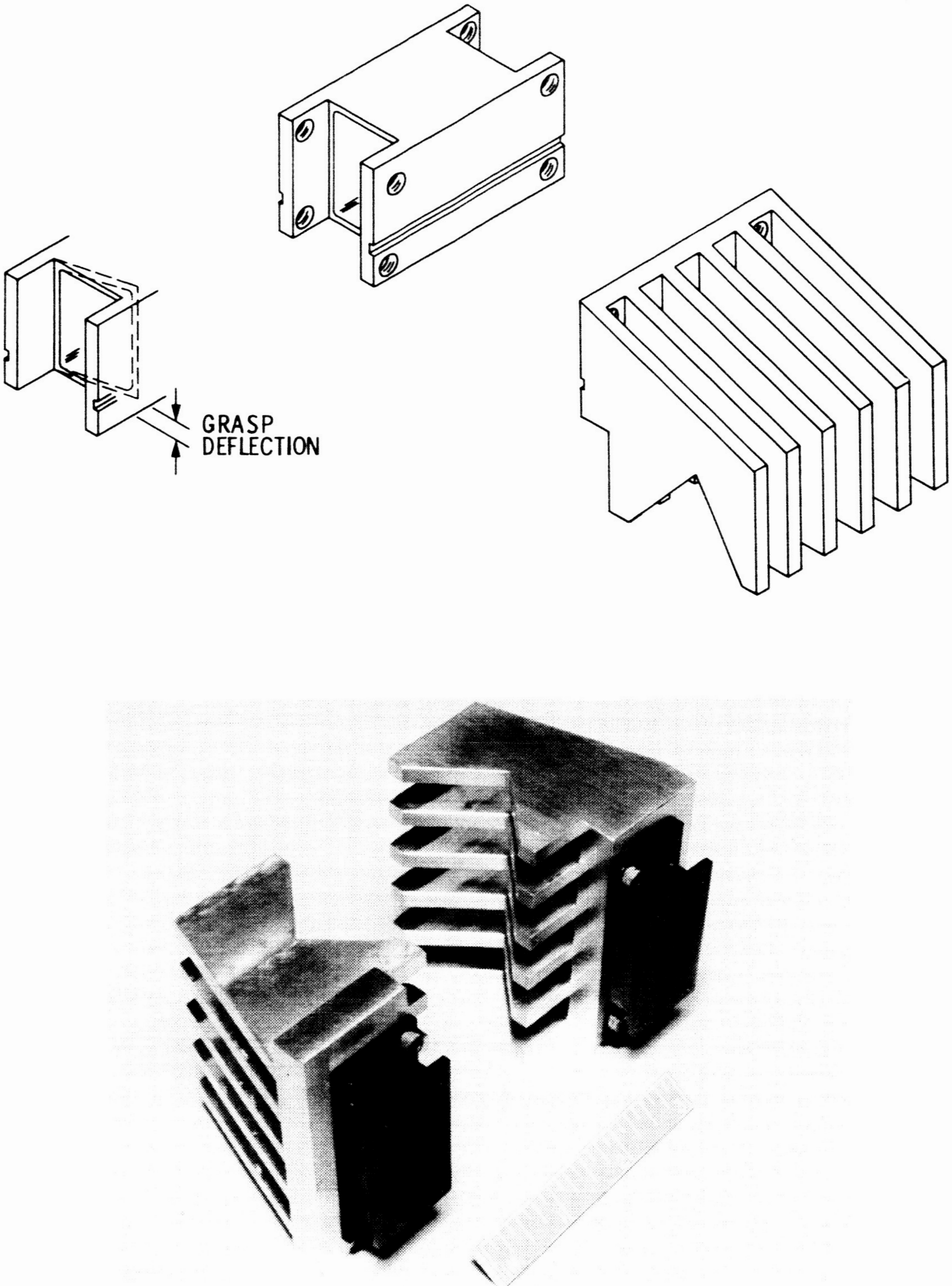


Figure 7. Intermeshing Claws with Grasp Force Sensor

parallel at all times.

- o Simple, modular, element "spacer" type design allows that the sensor can be added or removed as needed.
- o Strain gage bridge output signals can be read and processed by the same interface and electronics as used by the wrist force sensor.
- o Strain gage bridges can be mounted entirely on the inside of the structure and potted in silicone rubber for a robust element.

Figures 8-10 show the present state of the smart hand mechanical hardware development.

VI. SENSORS

This section describes the sensors integrated into this smart hand, namely the force-torque sensor, clamp force sensor, tactile sensor, and optical proximity sensor. Increasing the type and amount of sensors increases the complexity of the data handling required. Overall data handling hardware and software has been designed to use only seven slip ring conductors but still accomodate efficiently all sensors described below.

A) Force Torque Sensor

The force-torque sensor is designed to be mounted between the hand and wrist. The mechanical frame houses a Maltese Cross configuration machined from a solid piece of 7075T6 aluminum as shown in Figure 8 and 9. Semiconductor strain gages are bonded on all sides of the Maltese Cross deflection beams and form full bridge circuits. This insures against loss of accuracy due to temperature drift (References 2,3). This set-up results in eight sensor readings which are amplified by instrumentation amplifiers, read by the Sensor CPU, and then sent serially to the Signal Processing CPU.

The Signal Processing CPU resolves this into three orthogonal forces and three orthogonal torques in the sensor reference frame by a 6 x 8 transformation/calibration matrix. The sensor in this smart hand is specified to measure three orthogonal forces up to 30 pounds and the three orthogonal torques up to 50 foot-pounds with a resolution of 1 part in 500.

B) Clamping Force Sensor

Two semiconductor strain-gage clamping force sensors are designed at the base of both fingers (Figure 7). This is especially useful for the task turning nuts and of accurately clamping the fluid coupling mechanism for in-orbit fluid replenishment of space platforms.

When issuing a command, the control computer specifies a maximum clamping force parameter. The Servo CPU servos the motor so that the clamping force is dynamically maintained until a new command is received. When a task of known clamping limits is being performed, hybrid control can

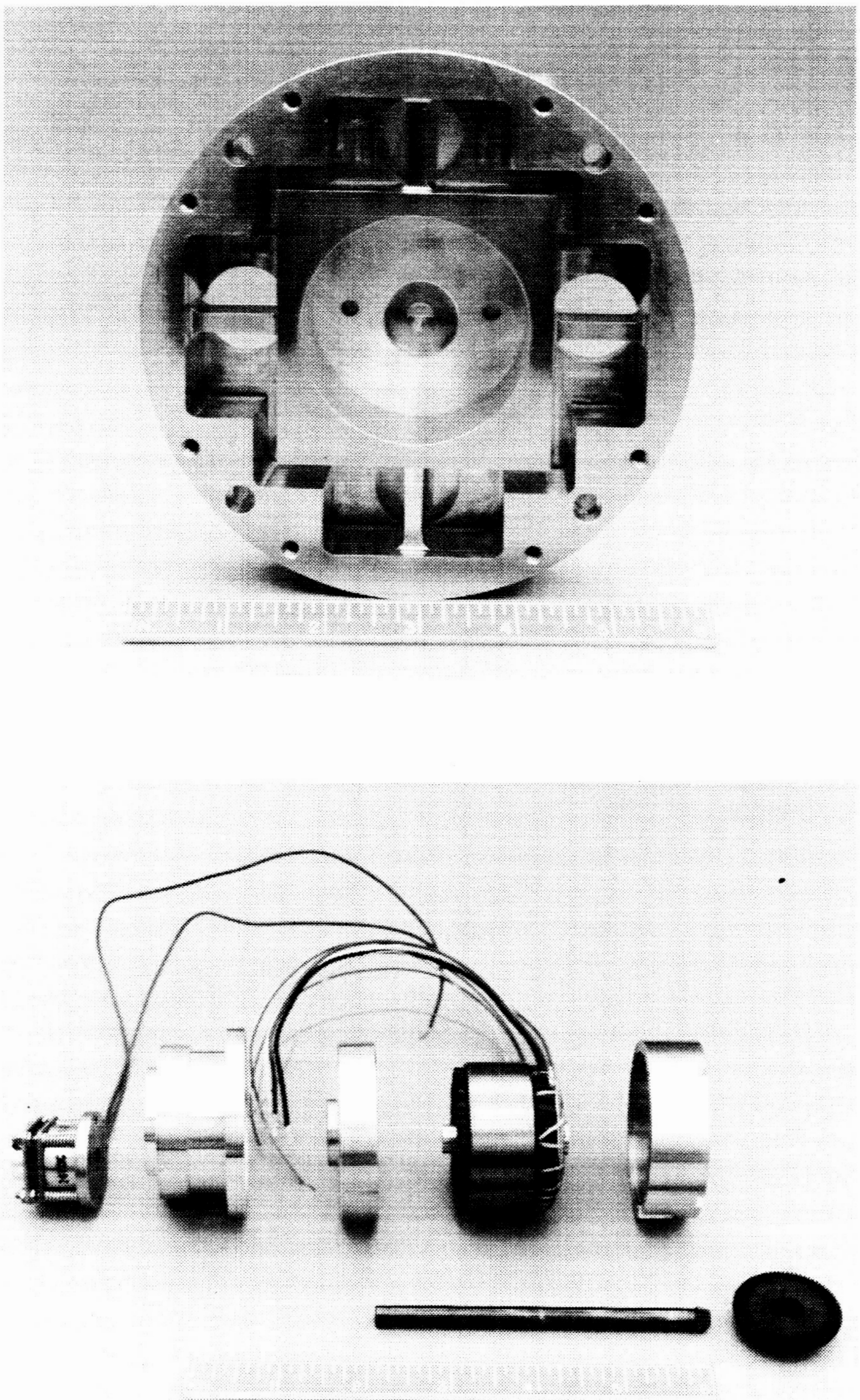


Figure 8. Force-Torque Sensor Frame Seen from Electronics Bay and Motor/Drive Components

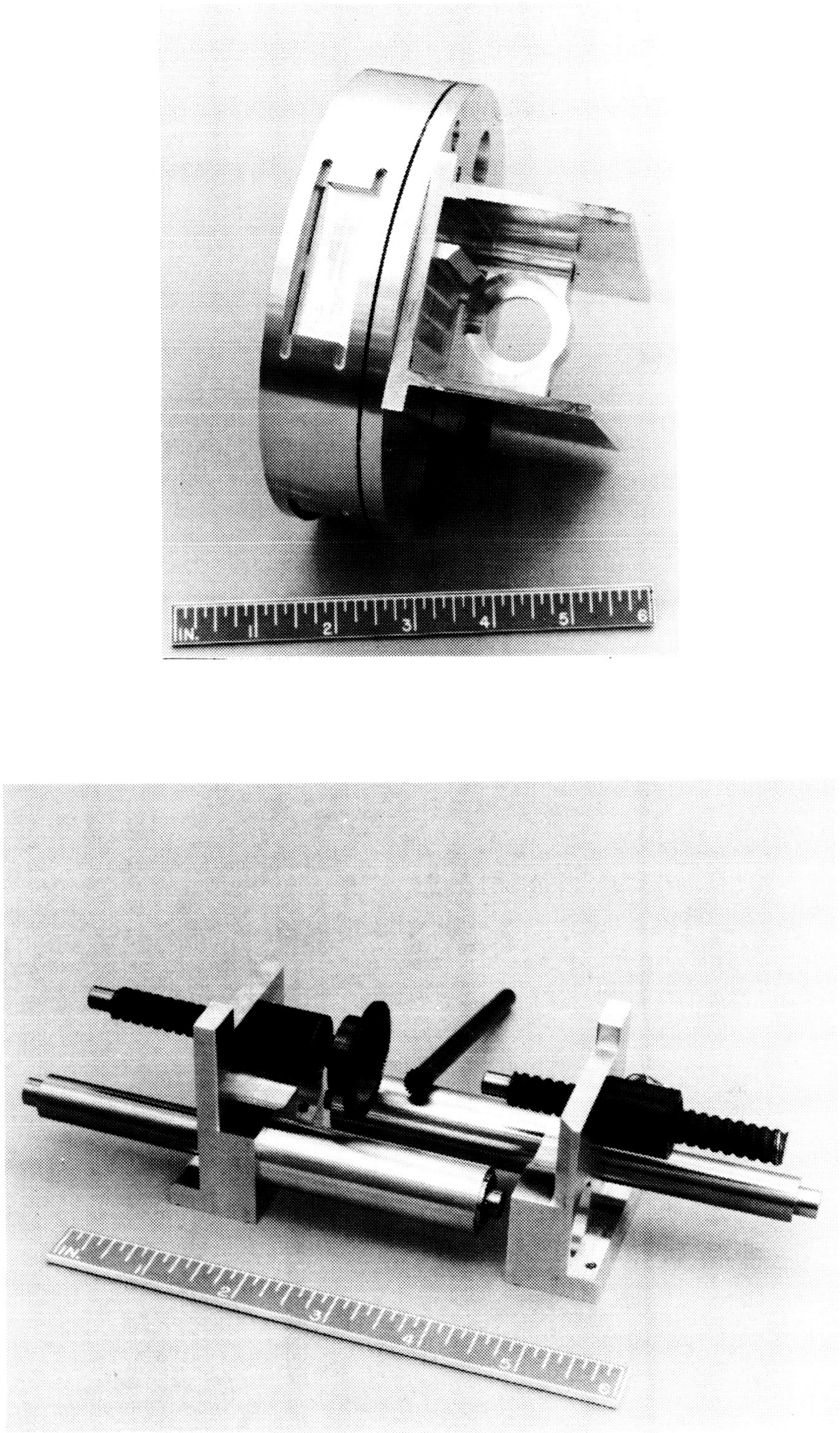


Figure 9. Force-Torque Sensor with Channel Frame and Drive/Support Column Components

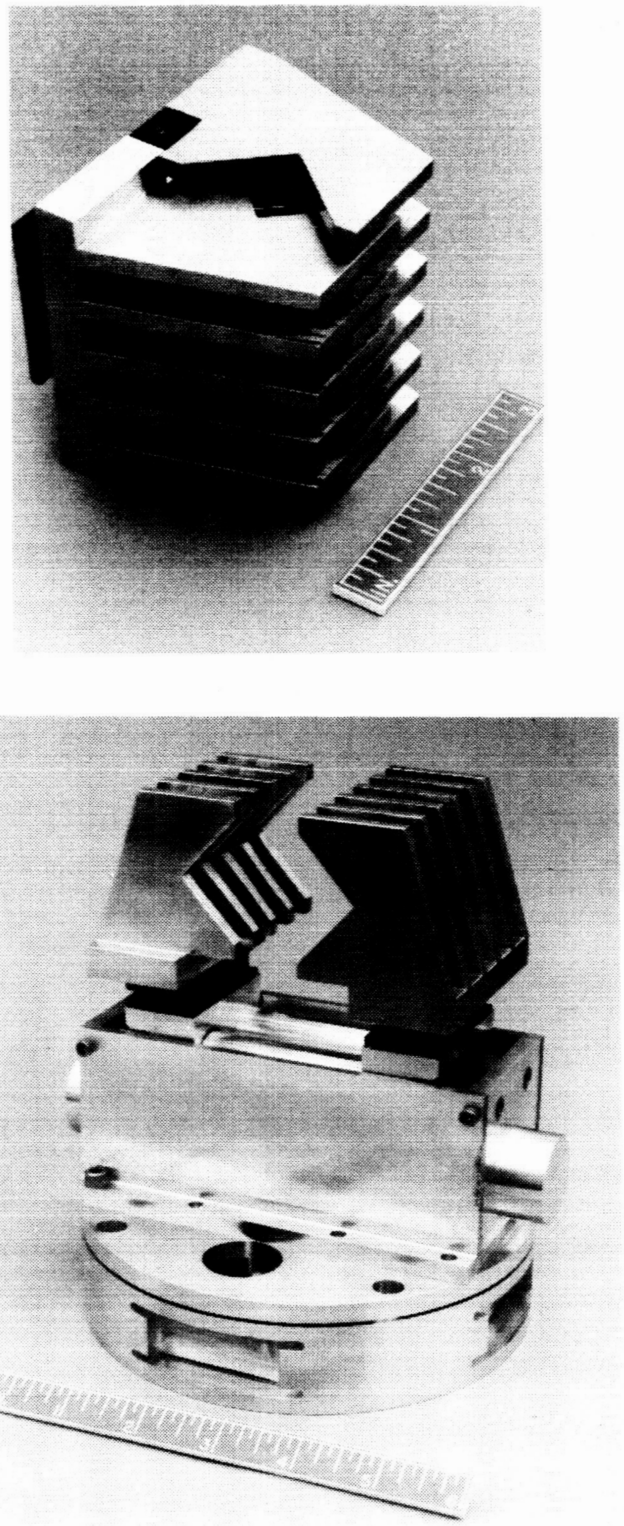


Figure 10. Hand Preliminary Mechanical Assembly and Claws Intermeshing

prevent operator-induced mistakes due to over-controlling or fatigue. The clamping force information can be used by the Servo CPU to override and prevent the operator from the extremes of crushing the object or from letting the object slip away.

C) Tactile Sensing

The data handling system has reserved 32 additional analog channels to accomodate area tactile sensing for each of the individual plates on the intermeshing fingers. This gives the shape or "force profile" of the object in the fingers and is most useful to detect misalignment of the object as it is being grasped. The misalignment and resulting torques in a zero-gravity environment could cause undesirable spin of the satellite being serviced.

D) Electro-Optical Proximity Sensing

Electro-optical proximity sensing will be installed in a future phase of the smart hand project. This sensing is desirable in a zero-gravity environment since the sensor information is available prior to physical contact. Capacitance or metal detection do not provide the generality or precision needed.

One proximity sensor consists of a photoemitter and a photodetector which are focused such that the optic axes of the two converge at a focal point. The distance of the object is derived from the intensity of the reflected light. To eliminate ambient light, the photodetector is first read with the photoemitter off and then read with it on and the first value is subtracted from the second.

The proximity sensors will be used to sense proximity both internally and externally to the grasp area. Four sensors shall sense forward with a range of 5 inches with a resolution of 0.05 inch. In this arrangement, the pitch and yaw orientation and distance can be determined and maintained. Four more sensors shall sense downward, such that in reference to a floor, the pitch and roll orientation and distance can be maintained. Internal to the grasp shall be eight sensors (four for each finger) so that the actual orientation of the object can be determined before it is grasped. The range will be approximately 1 1/2 inches.

VII. ELECTRONICS

A) Local Electronics

Three major drivers has stimulated the integrated design of the complex electronics that provide the multi-functional and high-level control of this smart hand. First, multiple sensor computer control and data acquisition at high sampling rates is needed. Second, transmission of data and power over a common rotary joint, via slip rings is to be implemented. Third, simplicity, compactness of design and minimization of power is always a basic requirement for space system.

The design of the local electronics incorporates a distributed microcomputer architecture, using advanced integrated circuits, including hybrid and high level multi-functional monolithic packages. This makes it possible to minimize the total chip counts, which are mounted on custom designed printed circuit cards. Such a card is shown in Figure 11. A single-chip data acquisition system is used as a front end-driver that performs multiple analog signal multiplexing, amplification, sample-and-hold, and finally analog-to-digital conversion. A single-chip microcomputer having its own serial data input/output port is chosen as the heart of the communication process. In fact, two of these are used in the present design. A third microcomputer chip actually performs the function of real-time motor control and sensor turn-on/off control (See also Reference 4).

As shown in Figure 3, the "local" (i.e., situated at the hand) electronics is composed of two subsystems, the sensor subsystem, and the end effector subsystem. Interfaced through the slip ring subsystem, this local electronics communicates sensor data, control computer commands, and receives power from the control station, which is remote from the end effector. There, three subsystems are configured - the signal processing subsystem, graphics subsystem, and the human operator control station subsystem.

As presently designed for the ground prototype of this smart hand, the sensor subsystem uses a 16-channel Datel HDAS data acquisition system. To accomodate a total of 27 channels of force-torque (8), proximity (16), clamp force (2) and position (1) readings, a separate multiplexer is used to access the proximity detector readings. Twelve-bit parallel data are shipped to a Motorola MC68701 microprocessor which then formats the data into a serial data stream to be sent through the slip ring subsystem to the Signal Processing computer. The same microprocessor drives the HDAS, selects the sensor data paths, and drives the proximity sensor emitter circuitry.

The end effector subsystem consists of two CPU's, the Motorola MC68701 and MC68705. The 68701 is basically used as a communication device by virtue of its serial input/output port; it also checks for transmission errors with a 16-bit check sum comparison. It receives the motor drive signals and control modes from the central computer(s) via the slip ring subsystem. This 68701 interfaces with the 68705 which stores and executes the program to control the 3-phase d.c. brushless motor. Pulse width modulation control and winding commutation control will be performed in this second CPU. It also receives the clamp force sensor, position and tachometer sensor readings for direct inside-loop control of the motor of the end effector.

The electronics is installed on one circular and two annular custom designed printed circuit cards behind the force-torque sensor, around and behind the motor.

Seven slip rings are used for interfacing the local electronics with

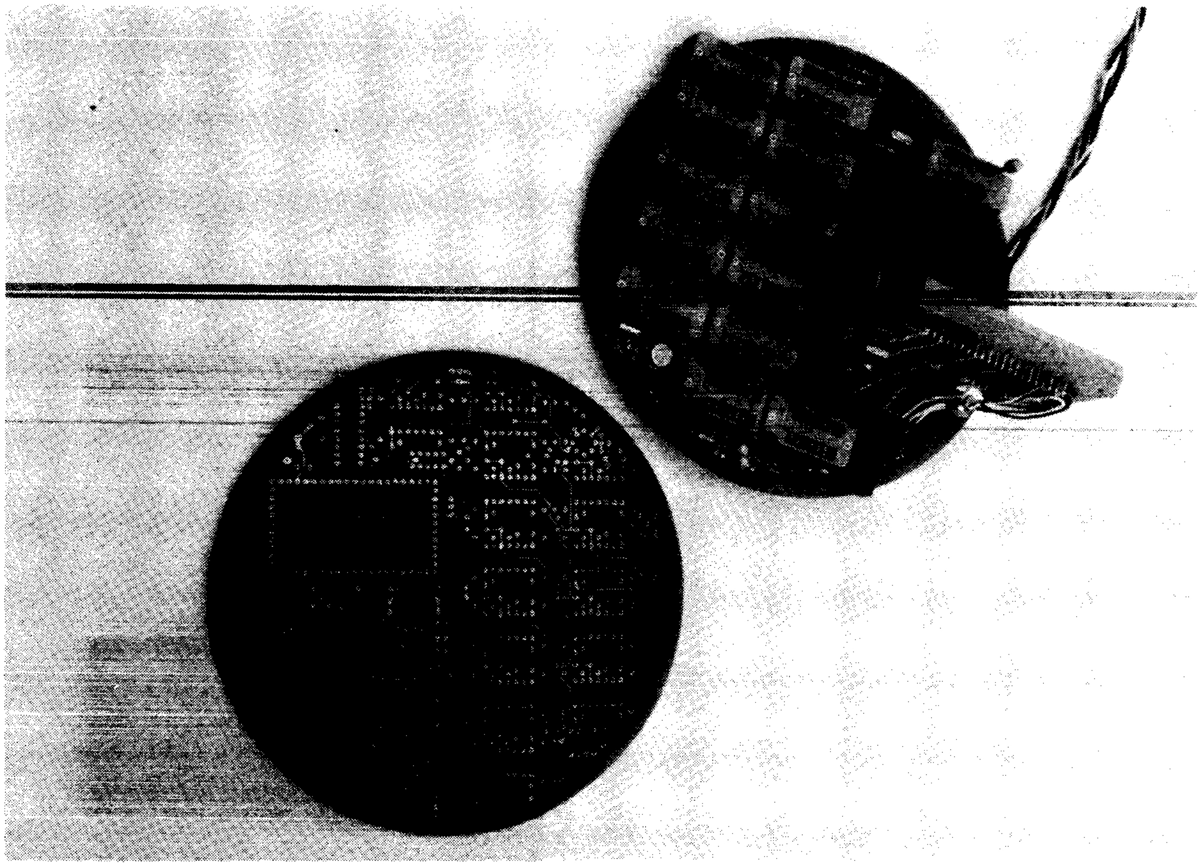


Figure 11. Part of Smart Hand Electronics: Several Microcomputers and Hybrid High-level Multifunctional Packages on Printed Circuit Cards.

the central electronics. Four will be used for power transmission, comprised of 24 VDC and 20khz 50 VAC. Two rings will be used for bidirectional data communication, one for the sensor subsystem, and one for the end effector subsystem. The last ring is for system ground.

B) Central Electronics

Since space, power and weight are far less at a premium here than at the end effector, most of the data processing functions and (naturally) the human interface functions are designed into the central electronics and control station. Again, Figure 3 shows the functional block diagram of this central electronics design.

Three subsystems and a power supply are the core elements of the central system: the Signal Processing Subsystem, Graphics Subsystem, and the Human Operator Station Subsystem. The signal processing subsystem will process the raw sensor data from the sensor subsystem (of the "local" electronics), including the conversion of the strain gage readings from the force-torque sensor and the clamp forces sensor into calibrated measurements through scalar and matrix multiplications. Proximity sensor readings and other readings are likewise converted to calibrated measurements. In addition, the Signal Processing Subsystem issues primitive graphic commands via an RS232 link to the graphics processor. Vector draw, area fill, and test insertion are typical commands. The graphics processor will then generate, at video rate, graphics pictures of the sensor readings and claw configuration on a TV monitor, to be presented to the human operator. The graphics processor is a Parallax 600-M-A unit.

This graphics subsystem is designed to generate video signals according to NTSC standards, and will be GENLOCK'ed to the local TV system. This enables the graphics display video signals to be compatible with standard ground based or spacecraft video monitor system nets. This also permits the mixing of video signals, the use of split screens, and the overlaying of one picture over another, which are important design features in a man-machine system.

Integral to this smart hand system are the displays, including the displays of the sensor readings and the displays/menus for the man-machine dialog. Sensor reading displays are designed to provide unambiguous, easy-to-interpret, convention standard, fixed as well as variable formats and presentation perspectives. Simulated displays of the hand approaching or closing in on an object will be presented, which will be of significant value to the human operator.

Finally, the Human Operator Station Subsystem will be designed to be user friendly, easy to use, and user interactive. Several levels of commands and menus will be provided, including system commands, display commands, sensor and motor control commands, and other interface and peripheral commands.

VIII. CONTROL

A) Position and Force Control of Smart Hand

The MC68705 microprocessor of the End Effector Subsystem is responsible for the inner loop control of the gripper. External commands are routed through the MC68701 microprocessor. These commands include gripper final position, x_d, maximum gripper closing velocity, x_{max}, and desired clamp force (at the gripper), F_d. Closed loop feedback data include the current position of the gripper, x_g, current gripper closing velocity, x_g, and current clamp force as experienced by the load, F_ℓ.

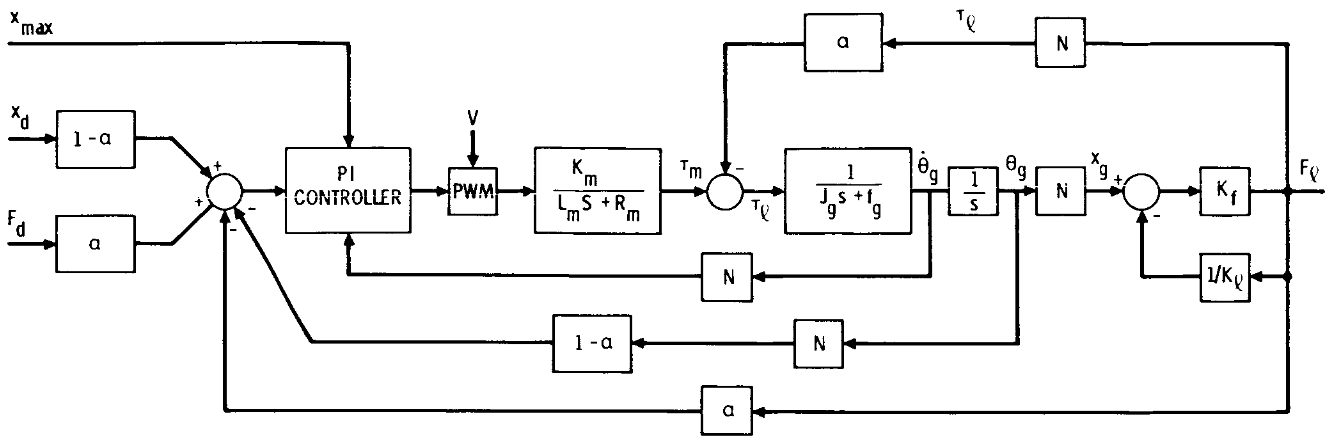
The MC68705 operates at a designed repetition frequency of 1 KHz. The pulse width of the motor drive pulse during each sample is determined in the previous sampling interval. In the present implementation, the motor is driven in a pulse width modulated mode rather than an analog current input mode in order to achieve better efficiency of the power supply. Commutation logic is implemented by this microprocessor.

Three modes of control are designed into the gripper closure, namely position mode, rate mode, and clamp force control mode. The position and rate modes transition into the clamp force control mode by the switching constant, α, which varies between 0 and 1. The block diagram of Figure 12 shows this hybrid control system.

In the MC68705, a new pulse-width calculation is acquired from the control program by the pulse-width-interrupt routine every 1 millisecond. The pulse width calculation is generated as follows: (1) Set motor drive pulse on. (2) Read x_g and x_d, as conveyed through the MC68701. (3) Perform the differencing operation $x_d - x_g$, and $F_d - F$. (4) Check against the preset deadband. (5) Perform the calculation in the PI controller. (6) Check for overflows. (7) Calculate the required pulse width (for the next sampling interval). (8) Check x_g against x_{max}, and turn off if necessary. (9) While processing Step 2 through 8, check for pulse-off-interrupt, as determined by the pulse width calculated in the previous sampling interval. Service this pulse off interrupt when flagged. (10) Set commutation logic, as required by the current motor shaft and phase angle, and by the direction of motor rotation. The above process applies equally to the clamp force control mode, using the appropriate force variables instead of position variables. Figure 13 is the flow chart of this software.

B) Interprocessor Communication and Control

The MSFC control station has a 6 degree-of-freedom joystick to control 6 of the manipulator's joints. All gripper command are typed into the Control Computer console by the human operator. The Control Computer sends commands to the gripper serially at 9600 baud through the slip rings to the Communication CPU, i.e., the Motorola MC68701 in the End Effector Subsystem. A six-byte record is used for all commands. The first byte of the record is the command byte which can be either a gripper move, gripper halt or a gripper status request. For a gripper halt or gripper status request, the remaining bytes are zero. For the gripper move command the



α = TRANSITION

x_d, $\dot{x}_{max}$, F_d = DESIRED GRIPPER OPENING, MAXIMUM SPEED, AND CLAMP FORCE

F_ℓ = ACTUAL CLAMPING FORCE ON LOAD/GRIPPER

x_g = OPENING OF END-EFFECTOR/GRIPPER

K_m, L_m, R_m = MOTOR CONSTANTS

J_g, f_g = END-EFFECTOR/GRIPPER INERTIA END DAMPING CONSTANTS

N = GEAR RATIO

K_f, K_ℓ = SPRING CONSTANTS OF CLAMP FORCE SENSOR AND LOAD

τ_m, τ_e = TORQUE FROM MOTOR, AND ON END-EFFECTOR

τ_ℓ = RESISTING TORQUE, TRANSLATED TO THE MOTOR

Figure 12. Block Diagram of the Hybrid Position-Force Control System

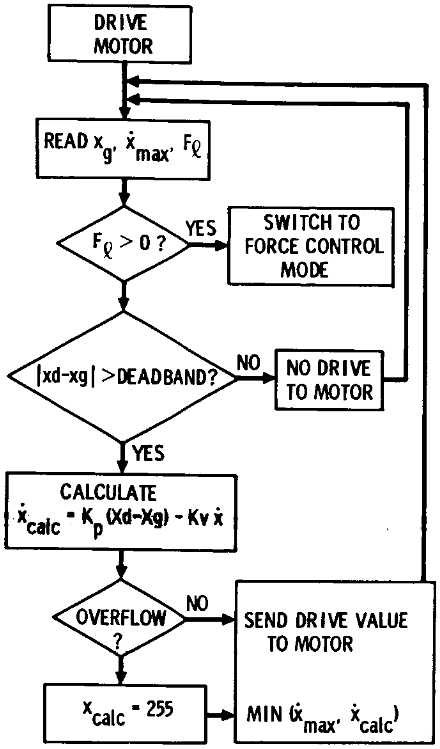


Figure 13. Flowchart of Motor Servo Control Microprocessor Software

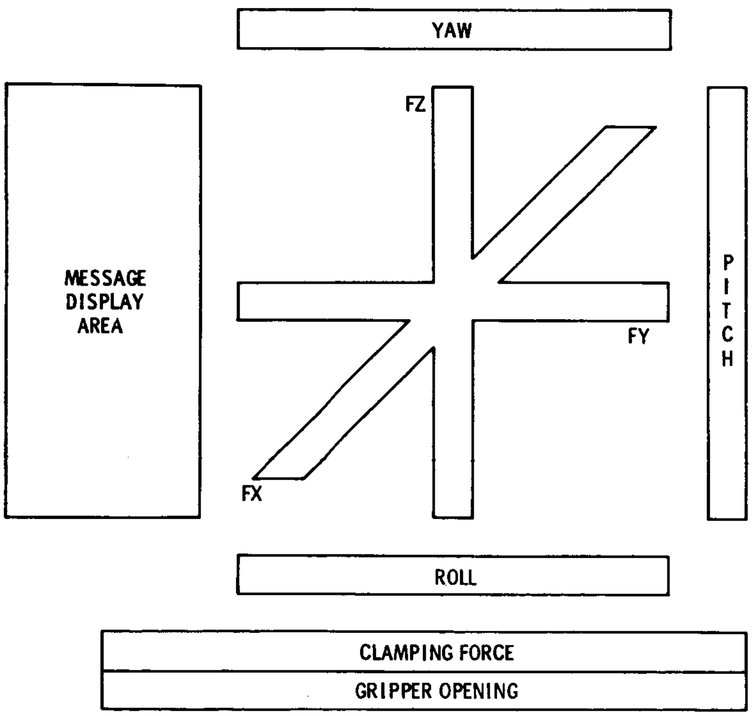


Figure 14. A Force-Torque Display

remaining five bytes contain the maximum gripper velocity, desired clamping force, desired final position and two checksum bytes.

Since the serial link is half-duplex, a hierarchy must be established for determination of communication direction. The Control Computer determines the directionality of communication over the Communication CPU. After a command is sent, the Control Computer changes the communication direction and waits for a reply. The Communication CPU computes the checksum of the record sent and immediately sends a message back if the checksum indicates a transmission error. Otherwise, the Communication CPU establishes handshaking with the Servo CPU (i.e., the MC68705 in the End Effector Subsystem) to inform it of the communication direction and sends to it the gripper command. The Communication CPU changes the communication direction and waits for a reply.

When the Servo CPU decodes the command to be a gripper status request, it reads and then sends it back to the Communication CPU. If the command is a gripper halt command, the brake is set, the motor is stopped and then the status is read and sent back. A gripper move command will servo the gripper in accordance to the velocity, clamping force, and position parameters and will read and send back status when completed. Note that in all three cases, the status is sent back to the Communication CPU, where a checksum is computed and the record is forwarded to the waiting Control Computer.

The Control Computer also determines directionality of communication with the Signal Processing CPU. The Signal Processing CPU determines the directionality of communication with the Sensor CPU. The normal direction is to have the Signal Processing CPU receive sensor data, except during the initial power-on communication test.

After initial power-on, the Control Computer sends test records to the Sensor CPU (via the Signal Processing CPU) and the Communication CPU. The purpose is to test the integrity of communication over the slip rings. If the test records are echoed back successfully, then the Control Computer will accept commands from the human operator.

IX. OPERATOR CONTROL

When the entire system is powered on, the Control Computer enters the Power-on-test mode and confirms that communication across the slip rings with the Communication CPU and Sensor CPU is operating properly. If there are communication errors, the test records will be incorrectly echoed back and no commands will be accepted from the human operator until the communication link is corrected.

After Power-on-test mode is completed, the Set-up mode is entered. The bias for the force-torque sensor is set and graphics options can be chosen. The color monitor can be set up for a split-screen display of several options. a standard video signal from a closed-circuit television camera may share part of the display. A three-axis coordinate system shown in a perspective view may be chosen to display the force-torque sensor readings (Figure 14). Bar graphs along the periphery of the coordinate system display the torques due to yaw, pitch and roll motions. Another display option will show the gripper opening, clamping force, as well as the optical proximity of the fingers to the object and external environment. The clamping force bar graph is displayed above the gripper opening bar graph with a different color. Depending on the task scenario, the two clamp force sensor data are displayed individually or as an integration of the two. All or just one of these options may be chosen to suit the task and operator preference.

The Normal mode consists of waiting for gripper commands to be entered, sending gripper commands to the Communications CPU and displaying the desired graphics options. The six-degree-of-freedom proportional joystick may be used to servo six of the arm's joints. The Set-up mode may be entered from the normal mode if there are no gripper commands presently being executed.

X. CONCLUSIONS AND FUTURE DEVELOPMENT

The smart hand project for OMV/PFMA applications and tests is carried out in three phases. Phase 1 will be completed this year and will include the whole mechanism, force-torque sensor, grasp force sensor, the related graphics displays, and hand closing/opening control. Phase 2 will be completed in 1986, and will include proximity sensing and control and related graphics displays. Phase 3 is scheduled for 1987 and will include some sensor-referenced automatic control functions interactively used with manual control.

The current smart hand development is based on an integrated design architecture, considering mechanism, electronics, sensing, control display, and machine interface in an integrated design approach. As conceived and breadboarded in this current design, this robot hand incorporates a state-of-the-art distributed microcomputer control architecture, utilizing advanced integrated circuits. Advanced multiple sensors are designed into this hand, leading to future implementation of shared manual and automatic control of the robot hand and of the robot arm. The implementation of the quasi-hybrid position and (clamp) force control will be fully evaluated once the electronics are fully built and integrated. Finally, the experimentation with the PFMA at MSFC will lead to results and conclusions on the actual control and human factor issues on the use of such a multi-sensor smart robot hand. It will also provide invaluable information to be utilized in the design of the next generation of hands, the multi d.o.f. smart, dexterous and anthropomorphic hands.

Acknowledgements

This work was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under contract to the National Aeronautics and Space Administration. Several people contributed to this project. The end effector mechanical design, including wrist force-torque sensor and grasp force sensor is by V. Scheinman of Stanford University. Ideas to electronics and data handling design were contributed by R. Dotson, R. Killion and H. Primus. Contribution of J. South to system description is appreciated.

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To be presented at the Satellite Services Workshop II
NASA Johnson Space Center, Nov. 6-8, 1985

DESIGN IMPLICATIONS FOR CRYOGENIC SERVICING OF SIRTF

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INTRODUCTION

During the next decade, the Space Infrared Telescope Facility (SIRTF), a one meter class, croygenically-cooled infrared telescope will be placed into earth orbit. The facility will offer a sensitivity improvement of 1000 over conventional ground based telescopes, and will examine in detail infrared sources previously catalogued by the Infrared Astronomical Satellite (IRAS). SIRTF is planned for a 28°, 2 year lifetime.

SIRTF will contain three infrared instruments, a spectrometer, photometer, and camera, requiring operating temperatures in the 2K range and mounted in a Multiple Instrument Chamber (MIC). As well, several mechanisms are employed. The secondary mirror drive provides chopping and image motion compensation. A beam splitter directs the incoming infrared energy to one of the selected instruments. In addition, a filter wheel, grating drive, and dichroics are included. The instrument heat load is on the order of 50 mW. An all superfluid, 4000 liter Dewar provides cooling. To maximize the scientific benefit of the facility, cryogenic servicing of SIRTF is planned.

Figure 1 shows a concept of the facility which is 8.45 m from end to end and approximately 2.85 m in diameter. The optical system is a 1 meter aperture Cassegrain with f/24 optics.

There are two options presently being considered for SIRTF. The first option is a free-flyer observatory with an MMS-based spacecraft. The second is a space platform mounted observatory, co-orbiting with the station. The free flyer concept is a 10.6 meter by 2.85 meter diameter telescope with solar arrays, high gain antennas and deployable appendages. A direct mounting to the STS (Space Transportation System) with integral trunnions is planned. For the space platform, only the 8.45 m by 2.8 m telescope will be mounted.

SERVICING CONCEPT

Figure 2 shows the SIRTF servicing concept for both the STS or SS (Space Station) facilities. The first servicing will take place in approximately 1995 with a two year replenishment cycle. In the case of SS based servicing, the STS will transport the ASE (Airborne Support Equipment) Dewar to the space station up to 2 months prior to the servicing need date. The Dewar will be stored in the refueling bay. From SS, the Orbital Manuevering Vehicle (OMV) will retrieve SIRTF from orbit and bring it to the servicing bay. The Mobile Remote Manipulator System (MRMS) will transport the ASE Dewar to the servicing bay, transfer lines will be connected along with power and signal umbilicals, and the aperture will be covered prior to replenishment.

The time required for preparation in the servicing bay is on the order of 6 hours. Figure 3 shows the hardware configuration for transfer. Changeout of the On-Orbit Replaceable Units (ORU) is accomplished prior to cryogen transfer operations if needed. ORU's are stored in a container with the ASE Dewar. Intra-vehicular activity (IVA) monitoring and control will be employed for cryogen transfer operations with ground monitoring occurring as well. Refueling of the OMV will precede hardline checkout of SIRTF from the IVA panel and ground monitors. At this point, there will be an Extra Vehicular Activity (EVA) to disconnect transfer lines and umbilicals, and remove the aperture cover. The OMV will berth with SIRTF and return it to the 900 km orbit. There will be a remote checkout of SIRTF on-orbit and if needed, a contingency return to the SS. At the first available opportunity, the STS will return the ASE Dewar to the ground for refill.

MECHANICAL/THERMAL INTERFACES

The hardware elements required consist of external and internal ASE kits. The external kit consists of the Dewar with pump, control, and monitor electronics, transfer lines, and the electrical umbilical. The Dewar is 4.34 m diameter by 1.78 m depth. The mass of the Dewar is 3360 kg full and 2355 kg dry. It contains 6600 liters of superfluid helium (SFHe) for transfer to the 4000 liter SIRTF tank. The internal kit consists of the command/data or command/data software only.

The mechanical interfaces required on the ASE Dewar are STS compatible trunnion mounts (2 or 4 sill, 1 keel) and 2 RMS/MRMS grapple fixtures. For SIRTF, STS compatible trunnion mounts (4 sill, 1 keel) are also required plus an FSS cradle A' mount on the aft end and 2 RMS/MRMS grapple fixtures.

Considering the loads environment, acceleration levels should not be a problem if no large changes occur during cryogen transfer. Large changes in g-direction or magnitude may require a temporary halt of transfer operations.

Thermal requirements for both the ASE Dewar and the SIRTF are an absorptivity/emissivity ratio of 0.2 -0.3. The SS "shell" temperature for both storage and transfer should be 300 K inside the tent. The tent is a solar shelter of beta cloth or multi-layer insulation (MLI) to reduce rapid temperature changes.

The total power dissipation in the tent during all operations should be below 0.5 kW to maintain temperature of the Dewar shell at 280-310K. To assist in this, thermally isolated trunnion mounts should be provided. In addition, the SIRTF aperature should be in shadow to as great an extent as possible.

POWER AND COMMAND AND DATA HANDLING INTERFACES

Power is provided by hardline from the SS servicing bay port to the ASE Dewar only. During normal transfer, less than 100 W is required at 28 VDC with a peak requirement of 200 W. Power required for checkout is TBD but less than 1 kW. The control console electronics (internal SS) require 100 W. There is a potential concern of electrostatic discharge (ESD) pulses during connection.

Command and data handling can be provided by hardline, fiber optics, or RF. The ASE computer or SS multi-purpose applications console will be used. Estimated data rates are 8 bits/sec for monitoring during storage and 512 bits/sec during transfer operations. The command rate is 8 bits/sec. Automatic monitoring is required of the quiescent Dewar in the storage bay with tasks only in the event of alarm. Manual vs. automatic control of transfer operations is to be traded off. Ground monitoring should be included in the loop for data/command control and possible TV. Electromagnetic interference (EMI) requirements are to be determined.

STICCR STUDY TASKS

Both Ball Aerospace Systems Division (BASD) and Lockheed Missiles and Space Company (LMSC) performed a SIRTF Telescope Instrument Changeout and Cryogen Replenishment (STICCR) study. The period of the study was from 10/84 to 8/85. Study tasks consisted of:

Development of design requirements to allow on-orbit cryogen servicing and selection of the most feasible replenishment method.

Development of telescope design for instrument and mechanism change on-orbit.

Analysis of ground and on-orbit operations necessary to support on-orbit servicing, including development of operational sequences and timelines.

Identification of key technologies requiring demonstration to allow reasonable engineering confidence that on-orbit servicing can be accomplished, including schedules and cost estimates.

Development of system concepts for the airborne support equipment.

Analyses and tradeoffs pertinent to the implementation of cryogen replenishment and instrument changeout requirements.

STICCR STUDY RESULTS

Figure 4 shows the SIRTF SFHe system modifications required for on-orbit cryogen replenishment. A tubular heat exchanger 2.5 cm in diameter is attached at the cryogen tank mounting ring for increased gaseous helium heat transfer, and subsequent reduction of cooldown time. Large valve orifices of 2.5 cm diameter reduce the pressure drop for pre-cooling the system. A large porous plus vent provides flexibility during transient cooldown and flexbility in replenishment fluid temperature by permitting transfer of warmer than normal helium during the cooldown transient phase. High efficiency thermal joints maximize thermal conductance in the telescope and the MIC to effect rapid cooldown. The effect of this is shown in Figure 5. An aperture cover which has a low emissivity surface finish on the inside surface minimizes the heat load to the fore baffle. Finally, a short 2.5 cm diameter vent line increases the warm temperature fluid flow for assistance in precooling the tank.

Figures 6 through 9 address the telescope cooldown. In Figure 6, the important point is that liquid accumulation does not occur until the tank is cooled below the saturation temperature of the liquid helium.

Figure 7 shows the quantity of helium required for cooldown of the SIRTF Dewar system as a function of tank temperature. Note that the time required is for the Dewar system only. Transfer lines, pumps, etc. are not included. An additional 4000 liters is required for filling the SIRTF tank after cooldown, requiring from 4000 to over 9100 liters for replenishment of the SIRTF depending on starting temperature.

In Figure 8, the cooldown time and total cryogen quantity is examined as a function of the transfer rate and the thermal resistance discussed earlier at each of the two telescope locations.

Figure 9 presents timelines for cryogen replenishment in the case of SIRTF tank temperatures of 2 K, 150 K, and 300 K. The numbers above the boxes represent time in hours.

It can be seen from the material thus far presented that it would be advantageous to transfer cryogen to the SIRTF at 2K. Approximately a factor of 4 increase in time is required fro transfer at 150 K, while a factor of nearly 6 is required at 300 K. The requirement for transfer at 150 K stems from the situation whereby the SIRTF cryogen tank would be empty, but by pointing the telescope into deep space, a temperature of 150 K could be maintained. The requirement for transfer at 300 K is a result of the need for instrument changeout. Figure 10 presents a comparison of instrument changeout concepts. Changeout of either the entire MIC or selected instruments is possible with either warm (300 K) or cold (2K) changeout. Figure 11 shows a typical configuration for instrument access during changeout, while Figure 12 addresses the impacts both on SIRTF and on the instruments of changeout.

CONCLUSIONS

The present SIRTF configuration is acceptable for cryogen replenishment with modifications required to the manifold, valves, and external access. Addition of a vent line is required.

The SIRTF is servicable from STS or SS but probably not in situ unless the system is at 2K. Even at this point, in situ servicing is risky at the orbital precession rates specified. Instrument changeout at cryogenic temperatures is not advisable from a contamination and safety standpoint, and the current design for instruments and the Dewar aft end requires modification for access and modularity.

Instrument changeout on STS and cooldown are on a very tight schedule, whereas SS is an enabling capability for instrument changeout. A clean room is preferred even for warm servicing.

SIRTF TELESCOPE LONG LIFE CONCEPT

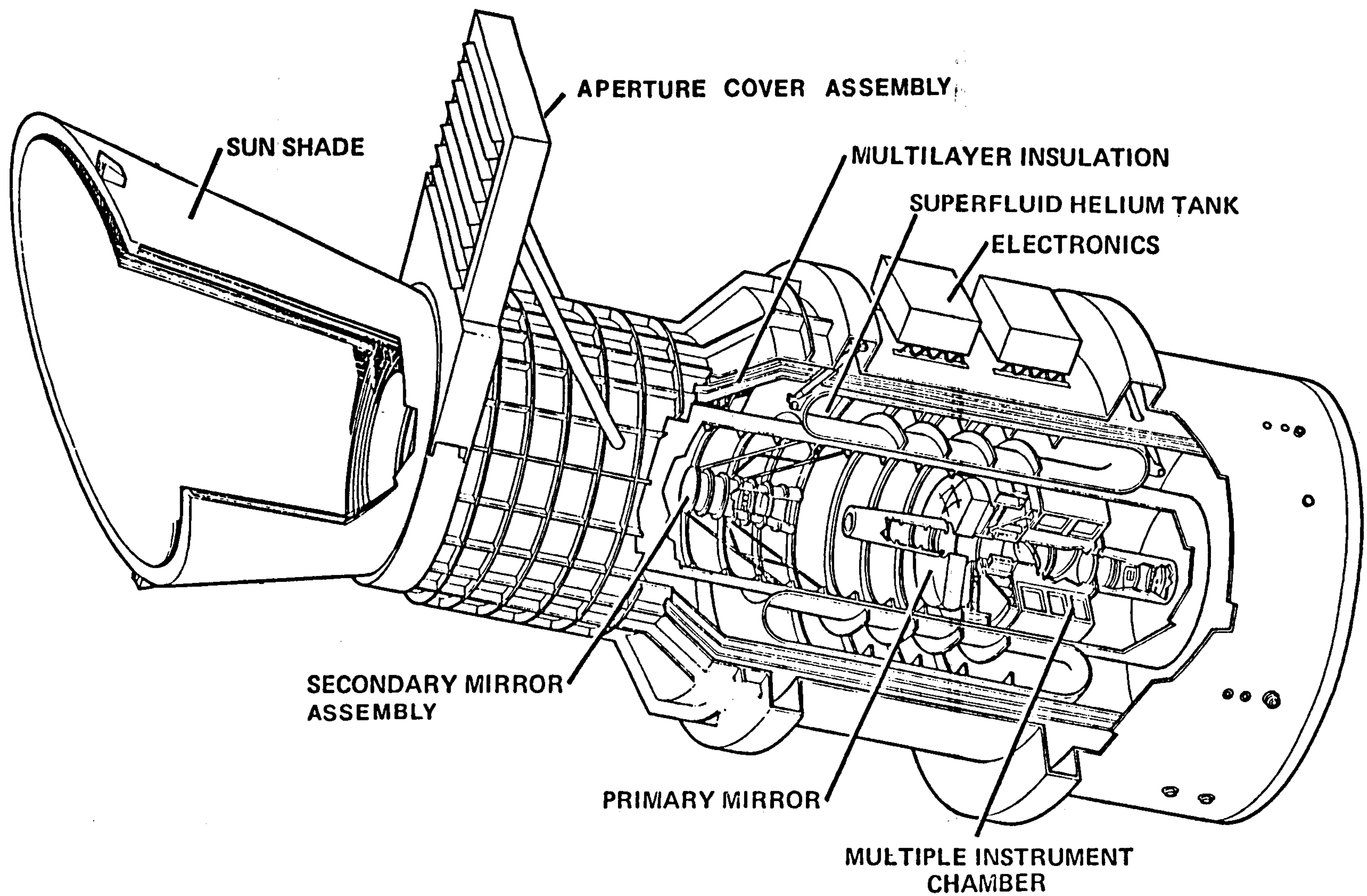
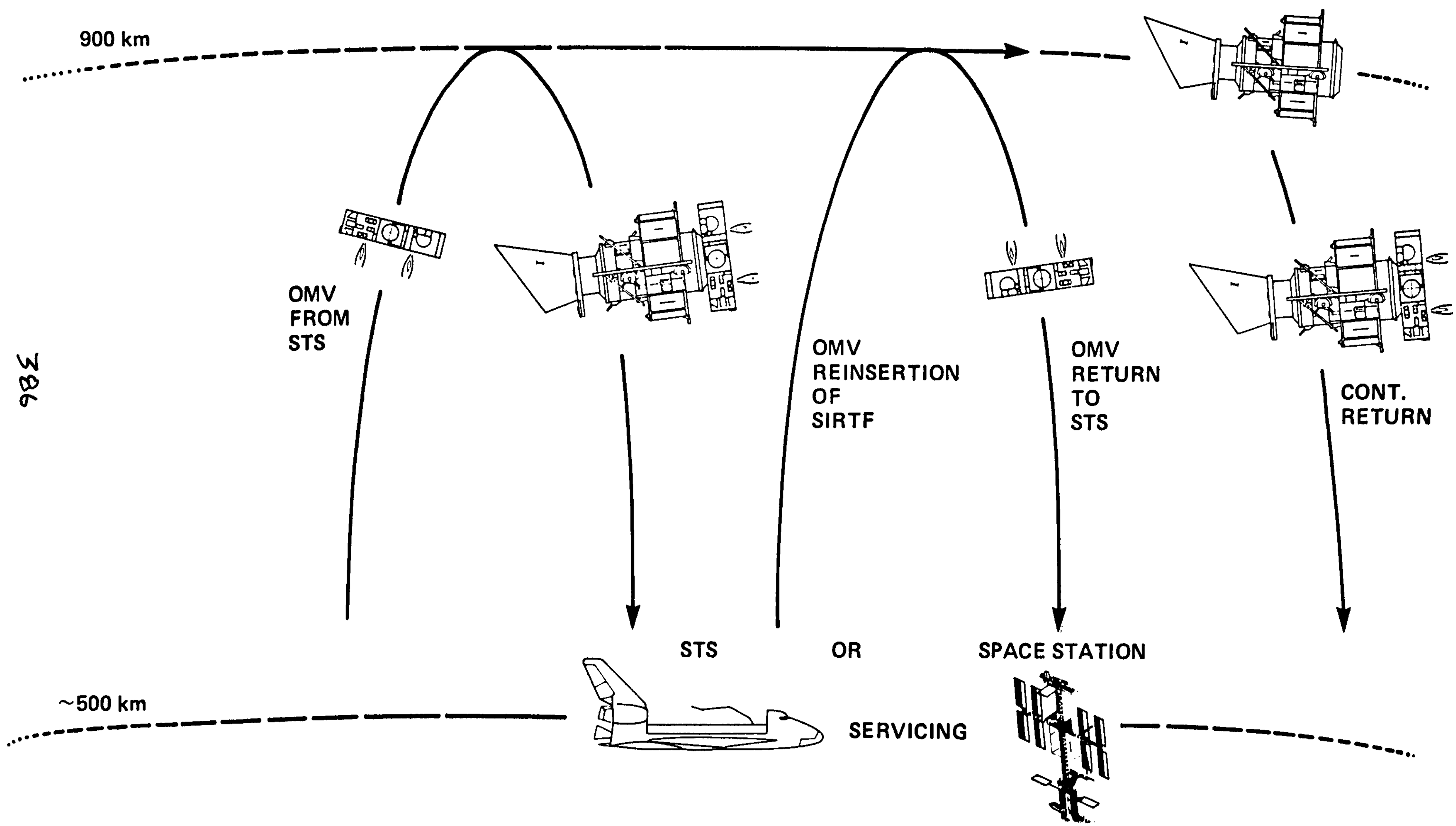


Figure 1

Figure 2



SIRTF RETRIEVAL, SERVICING, REBOOST, CONTINGENCY RETURN

OMV/SIRTF MISSION SCENARIO #2

HARDWARE CONFIGURATION FOR TRANSFER

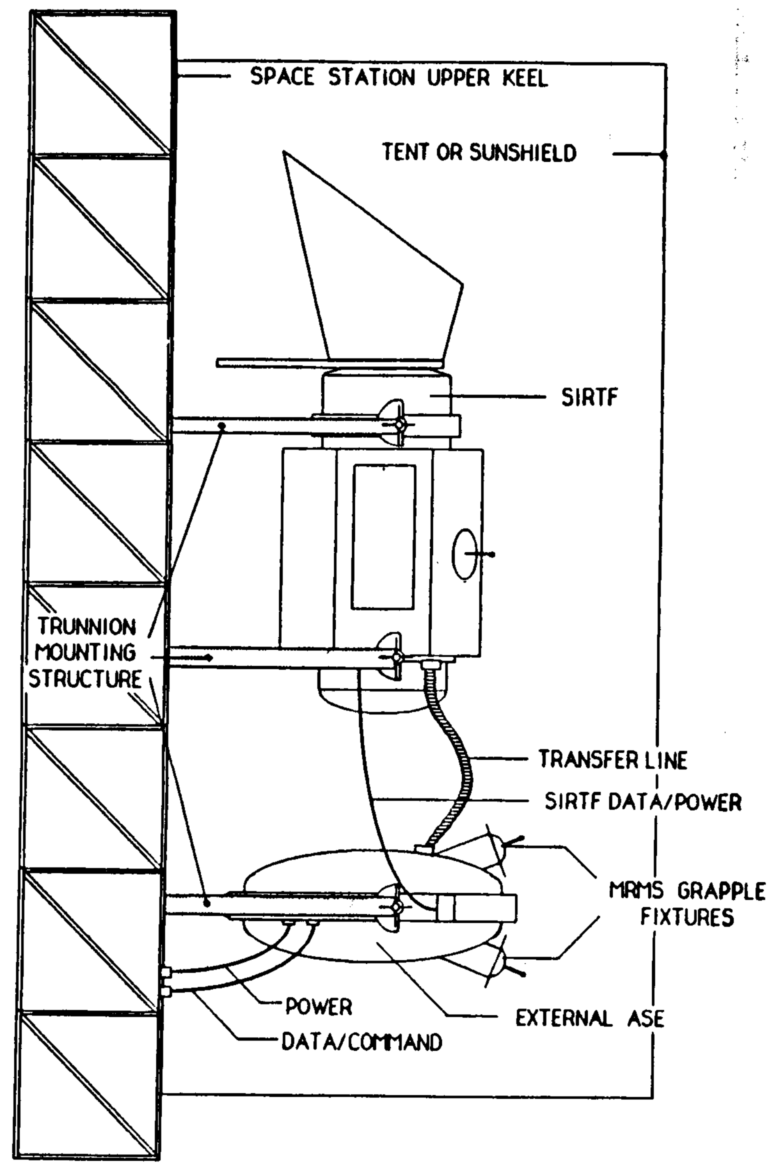


Figure 3

SIRTF SFHe SYSTEM MODIFIED FOR REPLENISHMENT

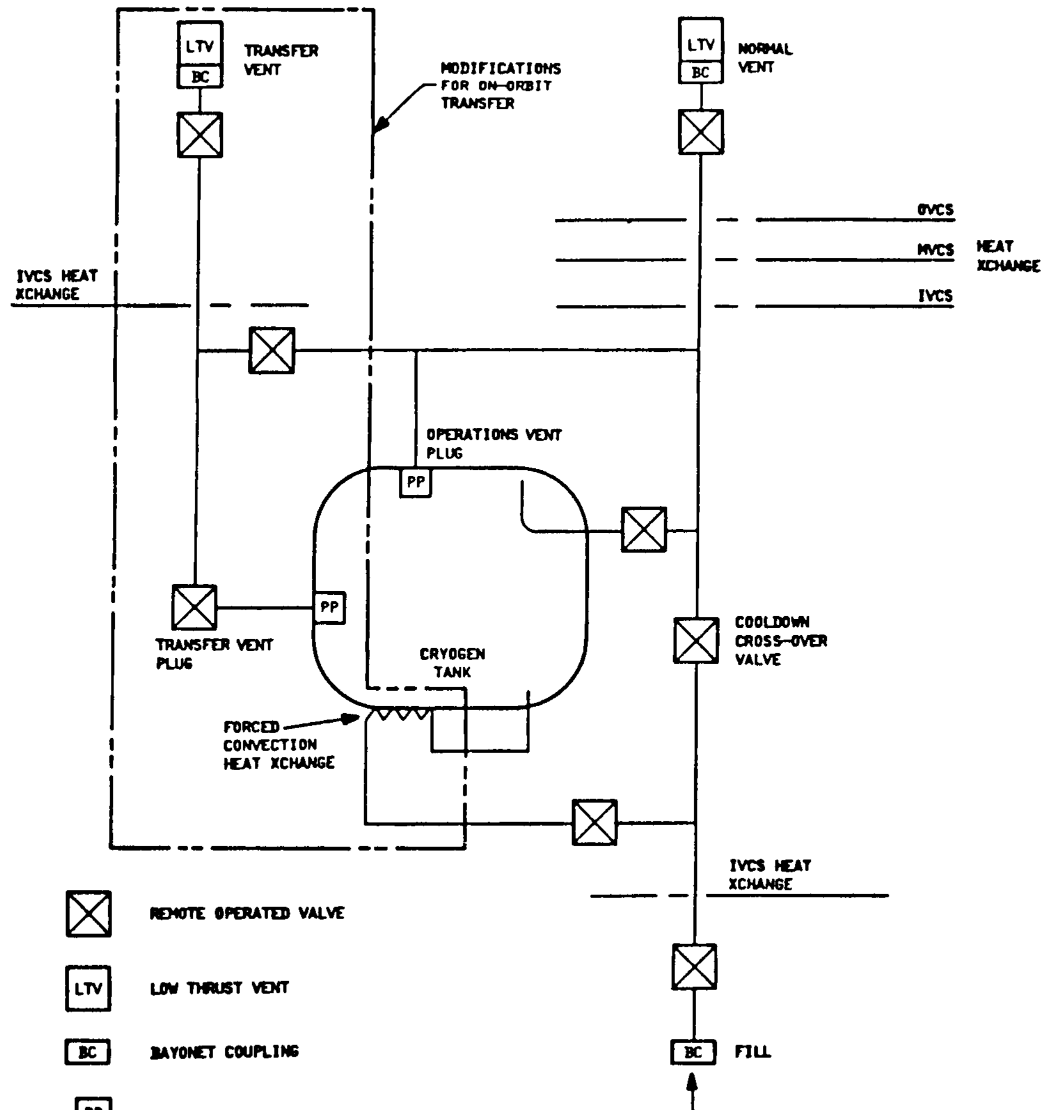


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Figure 4

TECHNIQUES TO REDUCE TOTAL HELIUM QUANTITY (2 OF 2)

- REDUCTION OF DEWAR/TELESCOPE INTERFACE RESISTANCE
  - LARGE INTERFACE AREA WITH HIGH-CONDUCTIVITY MATERIAL
  - DETACHABLE INTERFACE CAN BE SHRINK-FIT CONNECTION
  - UTILIZE CHANGE IN THERMAL CONDUCTIVITY WITH TEMPERATURE

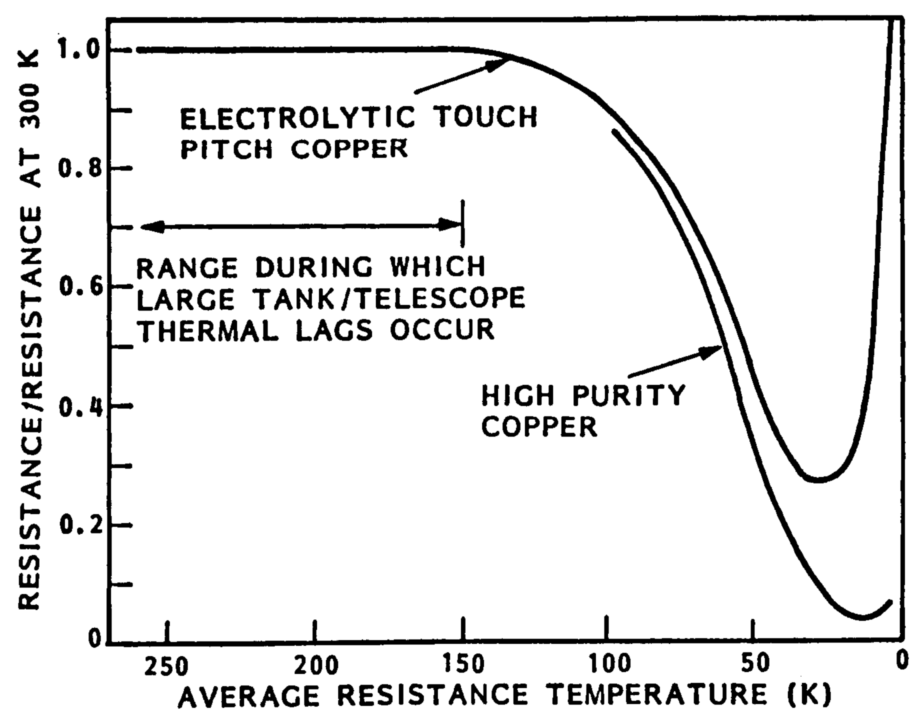

Figure 5

TELESCOPE TRANSIENT COOLDOWN (2 OF 2)

- CONSTANT TRANSFER RATE OF 100 l/h
- DURING THE COOL-DOWN, WHILE THE TANK IS ABOVE THE SATURATION TEMPERATURE OF THE LIQUID HELIUM, ALL THE TRANSFERED CRYOGEN WILL BE VENTED FROM THE DEWAR
- LIQUID WILL BEGIN TO ACCUMULATE IN THE TANK ONCE IT IS COOLED TO THE SATURATION TEMPERATURE, HOWEVER, HIGH VENT RATES WILL PERSIST UNTIL THE HEAT LOAD DROPS OFF. DURING THIS PERIOD, A SEPARATE CRYOGEN PHASE SEPARATOR IS REQUIRED TO PREVENT LOSS OF LIQUID

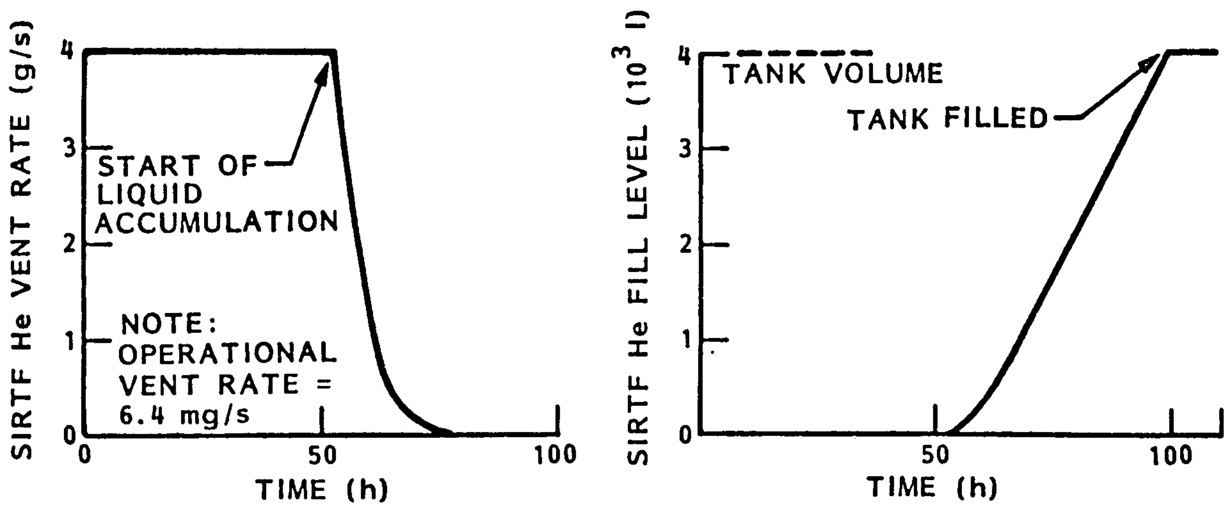

Figure 6

QUANTITY OF HELIUM REQUIRED FOR COOLDOWN

	SIRTF TANK TEMPERATURE		
	2K	150K	300K
HELIUM-II REQUIRED (LITERS)	0	3545	5078

Figure 7

A20AC5BU-8

TELESCOPE COOL-DOWN TIMELINE -CRYOGEN QUANTITY TRADE

- THE COOL-DOWN TIMELINE, TOTAL CRYOGEN QUANTITY, AND REQUIREMENT FOR A TOP-OFF ARE DETERMINED PRIMARILY BY THE THERMAL CHARACTERISTICS OF THE DEWAR/TELESCOPE INTERFACE AND THE TRANSFER RATE

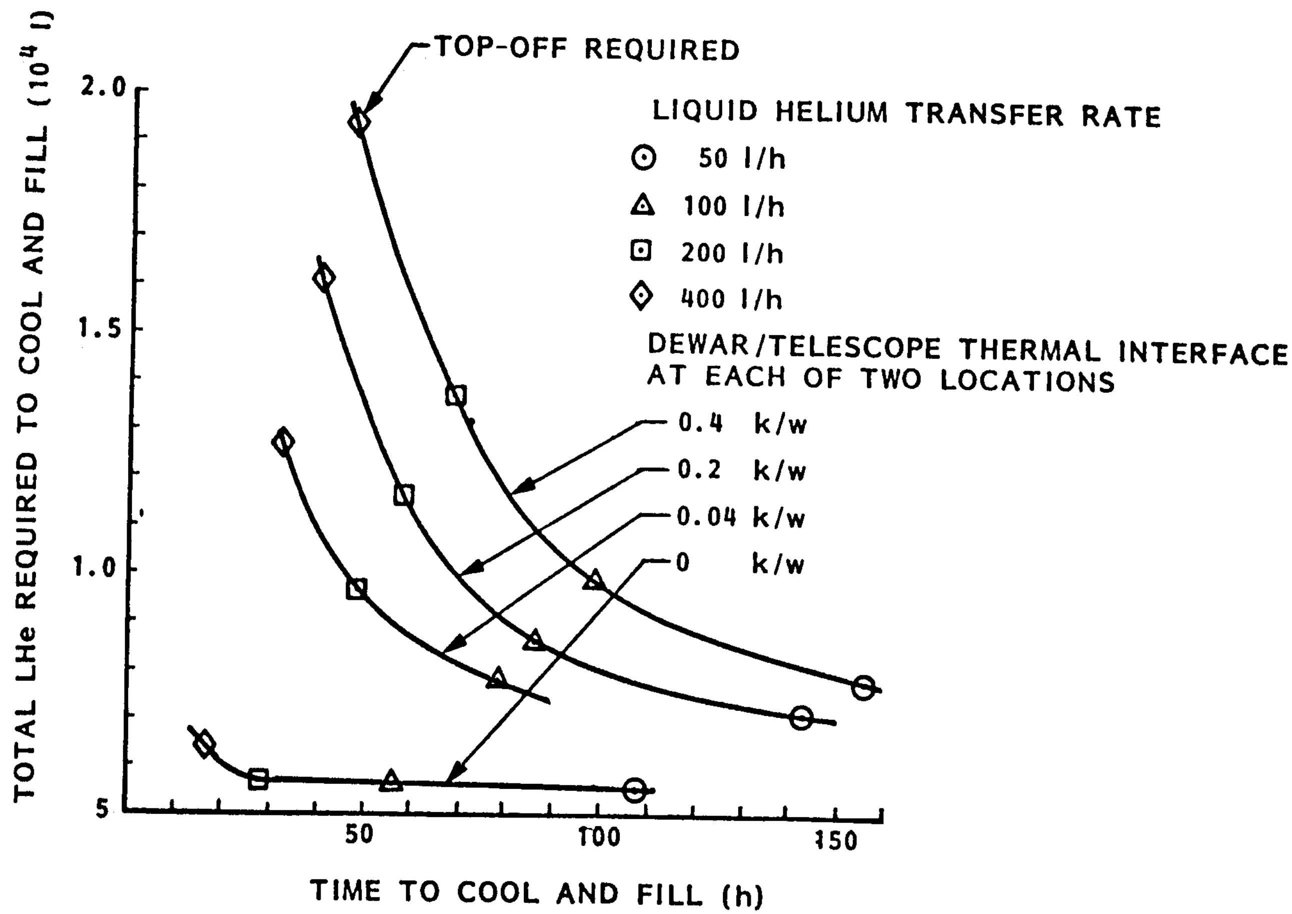

Figure 8. Telescope Cooldown Timeline

TIMELINE OPTIONS FOR SIRTF TRANSFER

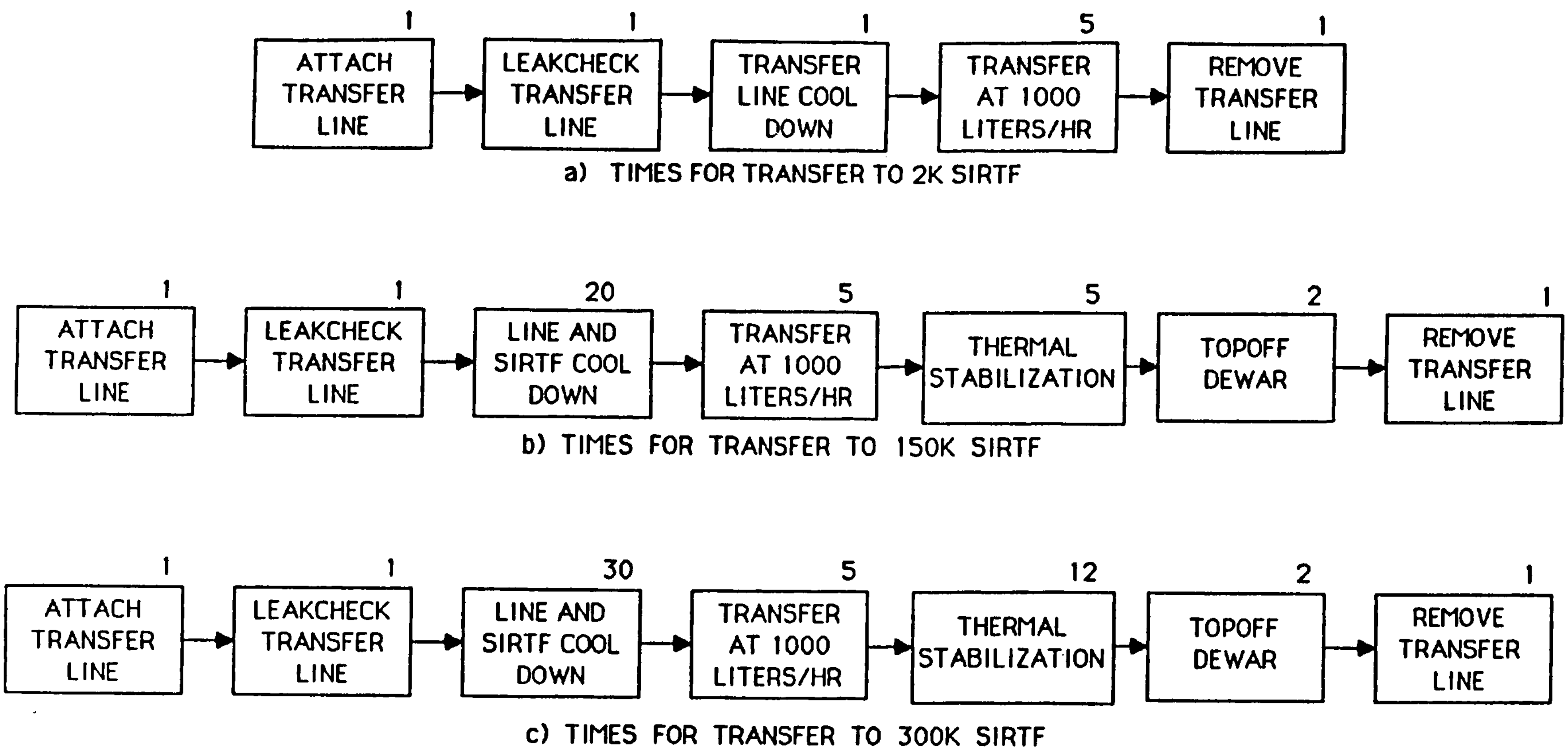

Figure 9

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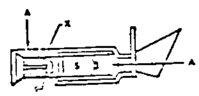
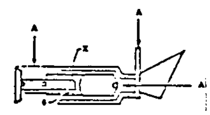
Figure 10

QUALITATIVE COMPARISON OF INSTRUMENT CHANGEOUT CONCEPTS

	WARM CHANGEOUT		COLD CHANGEOUT	
	1  Break in MLI/VCS at back end; MIC, telescope roll out for access	2  Break in MLI/VCS at back end; BS/FGS or instruments roll out individually for access	3 Entire MIC	4 Individual Instruments
Access for servicing instruments ● Structural mount ● Electrical connectors ● Beam path alignment ● 2-K connector ● 7-K connector	 1 1 1 1 1	 2 Requires long-handled tool 2 Requires device loop 2 Visual check of alignment not possible 1	 1 1 1 1	 1 1 1 1
Cryogenic mechanisms ● Fine guidance sensor ● Beam splitter ● Secondary mirror ● Cryogenic valves ● Aperture door	 2 May have to remove instrument for side access 2 1 1 1	 1 1 1 1 1	 1 1 1 Not possible 1	 1 1 1 Not possible 1
Remove entire MIC	1	Not possible	1	Not possible
Photon integrity	1	1	1	1
Contamination Control	2 Greater possibility of optics particulate contamination from rollers	1	3	3
7-K station and forward baffle cooling (vent gas routine)	2 Direct routine of vent gas not possible; conductive cooling through shrink fits	1	2	1
OMV, shuttle, or space station support points	1	1	3	4
PODS support complexity	1	2 Requires sliding collar, one end	3	4
Vapor-cooled shield attachment complexity	1	1	2	3
Cost	1	1	2	3

□ - Beam splitter (BS)
○ - FGS
▨ - Scientific Instruments
A - Access for servicing

O - 2 K station
X - 7 K station
S - Secondary mirror
C - Aperture cover
1,2,3 - Relative ranking
1 - Most desirable

INSTRUMENT REAR ACCESS

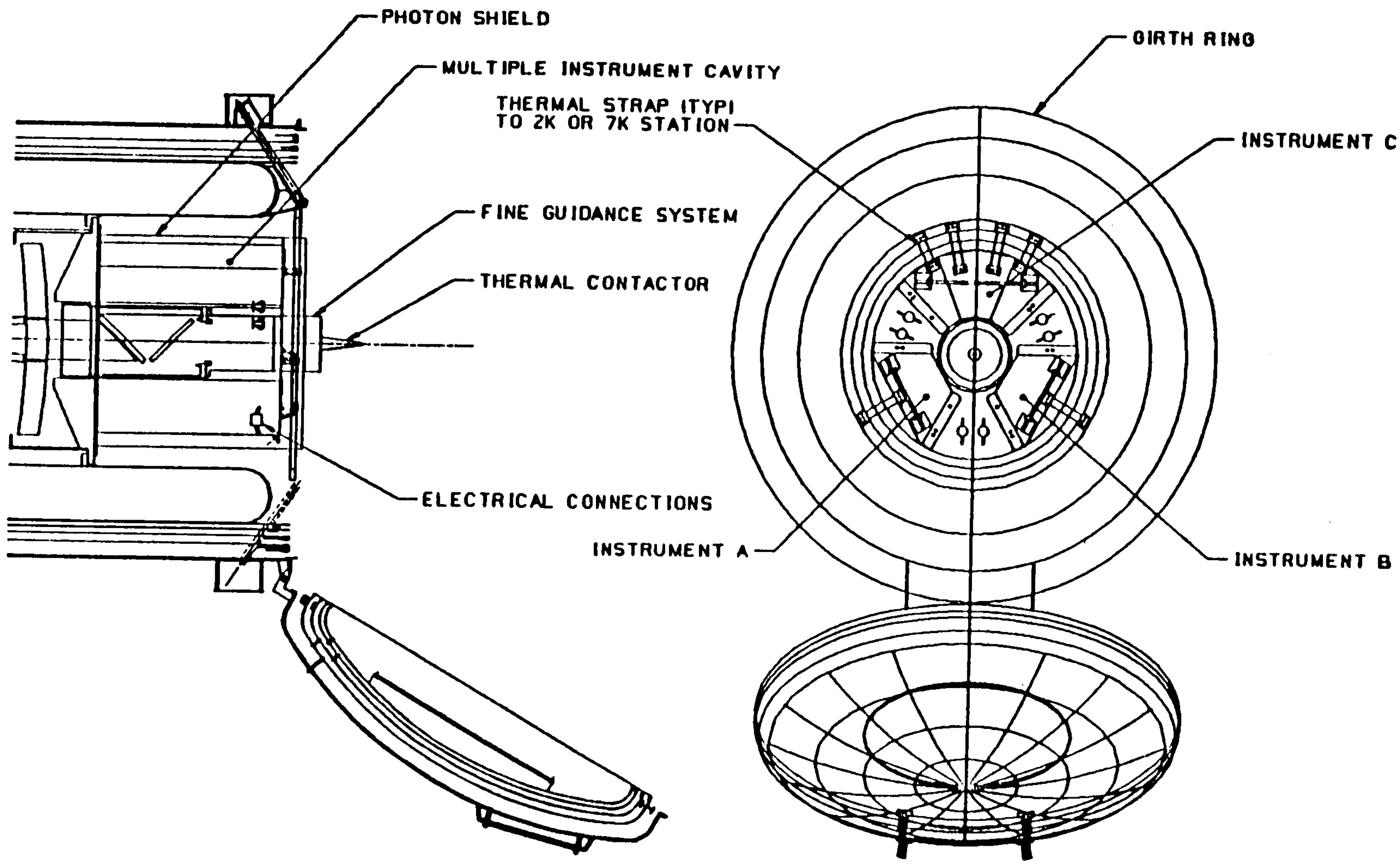


Figure 11

A/N 5708

IMPACT OF CHANGEOUT

ITEM	IMPACT ON INSTR	IMPACT ON SIRTF
GUIDE RAILS	REQUIRED	REQUIRED
THERMAL STRAPPING	REQUIRED	REQUIRED
HANDLES	REQUIRED	REQUIRED
ROUND CORNERS	REQUIRED	SOME
ELECT. CONNECTORS	LITTLE	NONE
WIRING	NONE	INCREASED

Figure 12

INSURANCE BROKER'S ROLE

IS THE SPACE INSURANCE MARKET ADRIFT ABOARD A DISTRESSED ROCKETSHIP?

Based on recent satellite disasters, insurance underwriters can easily relate to many of the popular science fiction movies. What they lack, however, is a Buck Rogers or Captain Kirk coming to their rescue. And rescue is certainly needed if they are ever to return to Earth.

Since the beginning of space activity in the private sector, the insurance industry has been a cooperative participant. Space insurance traces its roots back to 1965 on the early INTELSAT series and has continued through the recent disasters involving LEASAT 3, LEASAT 4, ECS-3 and SPACENET III. During all this time, the underwriters lost close to $460,000,000! Assuredly, this loss was not part of their intended flight plan. Until they can find a suitable rescue vehicle, it may prove difficult to book them on future missions.

Like any business, the insurance market expect a fair return on investment and effort. Unlike other businesses, however, insurers cannot precisely determine their product costs until after their product has been distributed and sold. This is in marked contrast with say, a satellite hardware manufacturer, who can reasonably and accurately determine the cost of the hardware, add an appropriate profit increment, and price the final product prior to sale.

Insurers' costs accounting methods can only estimates what the final product cost will be. If they guess wrong - they must bear the loss on that sold and delivered product. They can only hope to recoup their losses on any future sales.

This aspect of the insurance business is well recognized and can usually be dealt with easily. The anticipated casualty rate with automobiles, homes, factories, even human lives, can be actuarially studied. An actuary, using elementary statistical analysis can easily calculate a proper rating basis. In fact, the more individual exposure units studied, the simplier the task.

Unfortunately, the actuaries are not much help when they look at the space business. There simply is not an adequate number of exposure units to make a sound prediction. During 1985 and for the foreseeable future, no more than 25 or 30 satellites will be launched each year. Predicting how many of the millions of automobiles on our highways will crash this year is not difficult. Predicting how many of the scheduled launches will fail is impossible. Statistical analysis also requires the study of homogenous units. However, the significant variation in satellite design make such study impractical.

In developing a proper rating basis, actuaries also need to study the possible impact of an individual exposure unit's loss. In the case of an automobile, factory, etc. loss of an individual unit has minimal impact on the total writings in that category. This is certainly not the case with satellite insurance. Today's satellite owners are requiring insurance in excess of $125,000,000 per satellite. Using today's average rate levels, a loss of just one satellite would probably wipe out one half of the anticipated premium during a given year. During 1984, the total launch insurance premiums paid were approximately $150,000,000. The three notable 1984 losses totalled almost $300,000,000! So far, 1985 has produced in excess of $360,000,000 in losses.

It can be argued that Aerospace technology is advancing rapidly, and that failure rates will be lower in the future. While this may be true, the benefit is offset since the amount of insurance placed on each satellite is greatly accelerating. In the late 1960's, a satellite owner looked to place approximately $10,000,000 or $20,000,000 on a given satellite. Today that same owner might require as much as $125,000,000 worth of protection.

How then, do we solve the underwriters' problems, as indeed we must, if we are to expect continued particiption by the insurance sector in the space business? The problems must be solved; certainly no financier, stockholder or board of directors is prepared to commit the massive sums required for space activity without significant insurance protection.

We believe that a series of initiatives could improve the current situation:

* Satellite manufacturers (and sub-contractors) need to fully appreciate their customer's risk management concerns. If a satellite buyer will not be able to secure adequate insurance, there may be a need to increase or improve upon product warranties. Any application of new

technology that significantly advances the state-of-the-art, must be considered, since this technology will impose potential new risk on an insurer. The total sharing of information should be standard practice.

* Satellite owners and operators need to recognize the interests of the insurers. Early and comprehensive involvement with insurance specialists should be an integral part of the planning of any satellite launch. Some classic risk management techniques (risk transfer, reduction, avoidance, etc.), employed very early in the planning phase may significantly minimize the need for Insurance. Insurance concerns should be addressed as early in the game as possible, certainly well before any contracts are written with hardware manufacturers or launch agencies. With all of the red ink that was recently experienced by insurers, some aspects of the contemplated programs may be "uninsurable". These aspects need to be recognized early so they can be modified or deleted.

* Despite the tremendous losses, insurers need to maintain a positive underwriting attitude. Increasing rates is only one solution for an underwriter who is faced with an unprofitable Loss Ratio. There are other techniques that may need to be explored. These would include: coverage changes, different pooling arrangements to spread risk, possible increased deductible application and co-insurance. No one doubts the insurers right to earn a profit. However, the return to a profitable position must be accomplished with minimal negative impact on affected industry. Fixing rates much higher than today's may effectively force operators to seek risk transfer remedies elsewhere. Such action could divert premiums from underwriters and delay their return to profitability.

* There is an increasing move toward the "privatization of space." To the extent that private industry can serve its own needs, government need not be involved. To the extent, however, that private industry cannot cope with its problems, government involvement may be appropriate. The tremendous economic recovery exhibited by post-war Japan was possible because business and government worked toward common goals. Similarely, to the extent that government, (most notably NASA) can minimize risk for a satellite owner, it should do so. Much can be done by government simply by permitting more launch scheduling flexibility to satellite owners. This would minimize possible excessive exposure build-up on a shuttle launch.

Consideration might also be given to government indemnification for damage to a satellite while it is in NASA custody (i.e. aboard the shuttle). This would greatly expand the potential insurance market capacity.

THE BROKER'S ROLE

The space insurance broker sits between the parties in solving today's problems. This specialist must recognize and respond to the legitimate concerns of the manufacturers, the satellite operator and the insurers. It may be that today's method of placing an individual direct insurance program may not be the best course to follow in the future. The broker must consider and possibly apply other more innovative and creative risk management techniques. This could include: different risk-pooling arrangements for multi-satellite users, perhaps some combination of coverage that would include the interests of both the manufacturer and satellite owner; and possible shared salvage provisions which could benefit more interests and thus minimize the final loss.

Whatever the design of the insurance program, these programs need to be comprehensive and reasonably priced. A balance must be struck to protect the interests of all parties.

In the end, it is in the best interests of all parties to work together to solve today's problems. The solution does not lie in space, it is here. There needs to be a little bit of Captain Kirk in all of us to successfully bring the ENTERPRISE safely home.

MAJOR AEROSPACE INSURANCE LOSSES

YEAR	SATELLITE	INSURED VALUE
1977	OTS I	$ 29,000,000
1979	ECS	14,000,000
1979	SATCOM F-III	77,000,000
1982	INSAT 1A	65,000,000
1982	MARECS A	20,000,000
1984	WESTAR VI	105,000,000
1984	PALAPA B2	75,000,000
1984	INTELSAT 5	102,000,000
1985	ARABSAT 1A	25,000,000
1985	LEASAT 3	92,000,000
1985	LEASAT 4	94,000,000
1985	ECS-3	65,000,000
1985	SPACENET III	85,000,000
		$ 848,000,000

PRE-MISSION PLANNING INTEGRATION

FOR

SATELLITE SERVICING

Presented To:
Satellite Services Workshop II
NASA - Johnson Space Center
Houston, Texas
November 6-8, 1985

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ABSTRACT

A consideration of immense importance to the success of satellite servicing operations is presented. The development of a simplified, standardized planning methodology for integration and readiness activities is shown to be a factor in cost and risk reduction for future space programs.

Pre-Mission Planning Integration for Satellite Servicing

INTRODUCTION

As we move toward routinely planned satellite servicing with the Shuttle and Space Station, we find that new satellite program concepts have all included servicing as a natural means of extending useful life. Furthermore, these programs are all building on the successes of Shuttle servicing missions. Everyone in the spaceflight business, it seems, is now a believer in the benefits of satellite servicing. What is more, there is a groundswell of effort to establish satellite servicing standards, techniques, tools, equipment and facilities.

National space program activities are sharply focused on servicing design requirements, both for the spacecraft and for the on-orbit facilities needed to accommodate them. What we should take a moment to consider also, however, is the development of supporting pre-mission planning, integration and readiness activities. It is imperative that replacement equipment be tested and verified with a working model of the spacecraft prior to the servicing mission. By its very nature satellite servicing requires the careful dovetailing of pre-mission activities of two or more projects - the serviced and the servicing - to which can be added the myriad of supporting agency tasks.

It must be strongly emphasized that, given reliable systems, pre-mission planning integration is the key to successful satellite servicing. It follows that not only do servicing designs, tools and techniques need to be standardized and simplified; but the overall servicing planning and integration methodology needs to be standardized and simplified as well!

In consequence, we attempt here to explore the development of pre-mission planning integration methodology for the coming growth of satellite servicing.

REQUIREMENTS

A recent functional analysis was performed for an upcoming national space observatory. The four basic functions shown in Figure 1 include the Service function. Servicing is comprised of the four tasks shown in Figure 2. From these four tasks 91 subtasks depend. Fifty of these subtasks, shown in Figures 3 and 4, are generated from the basic Prepare and Maintain functions.

One can deduce from the functional flow analysis that satellite servicing has substantial pre-mission, or inter-mission, preparation to perform. Two important requirements surfaced for satellites to be initially serviced from the Shuttle, transitioning to the Space Station as it becomes available:

1. It is important that simple, available (when needed) tools and techniques which utilize STS experience and NASA program planning be utilized in servicing missions.

2. Preparation for servicing must include well-integrated logistics, mission planning, crew and ground support training, component test and verification with a working model of the satellite, and disciplined scheduling and control.

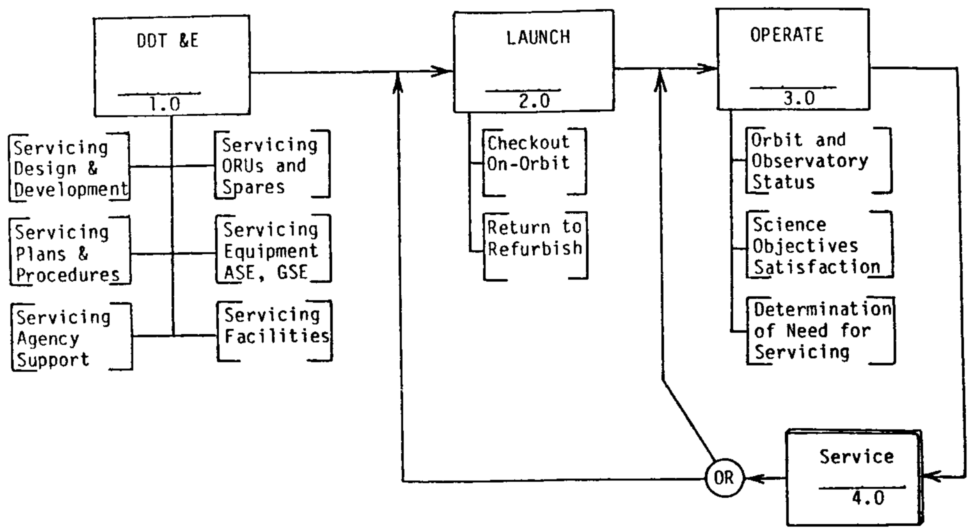

Figure 1 - Basic Observatory Project Functions

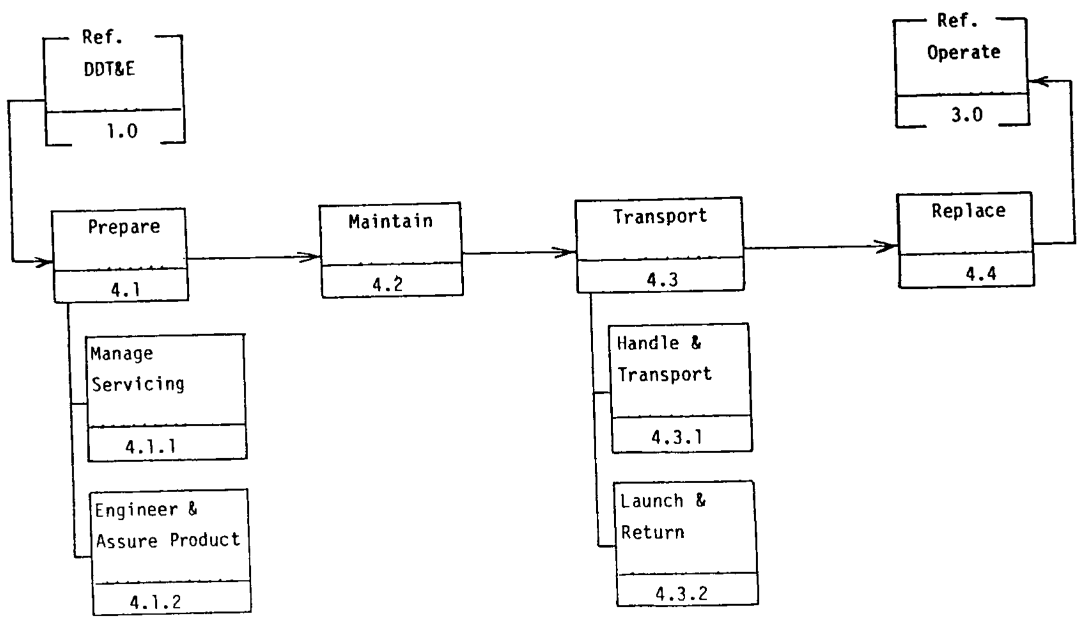

Figure 2- Observatory Servicing Functions

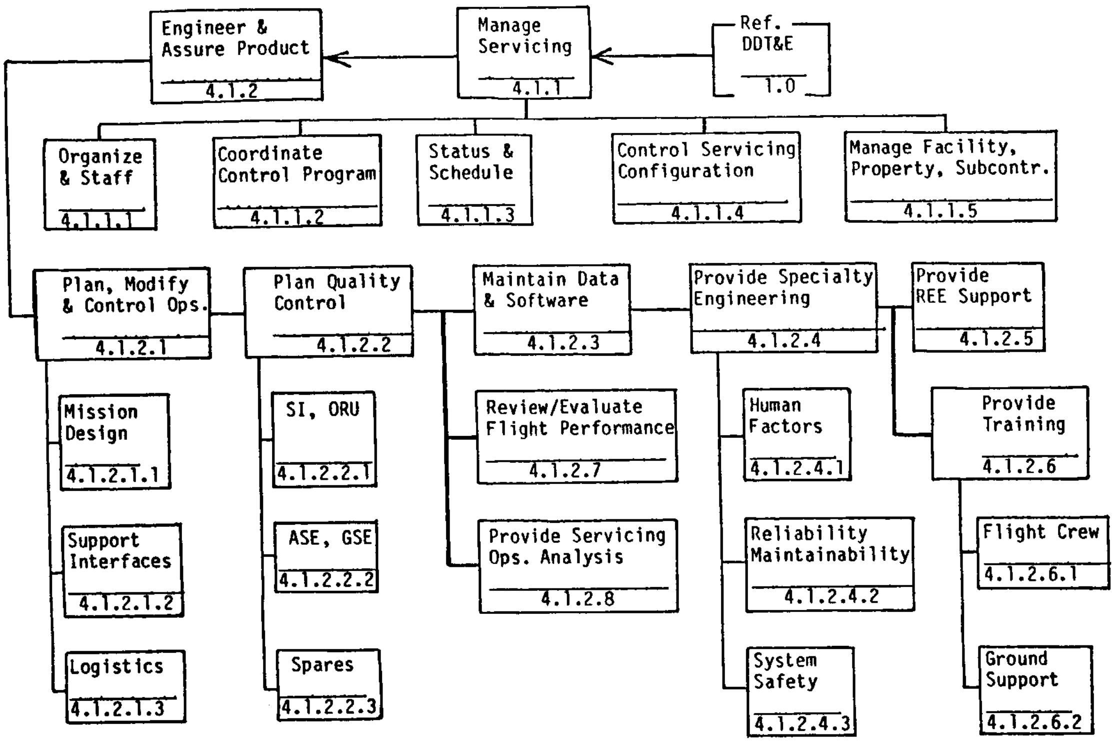

Figure 3 - Satellite Servicing Preparation Functions

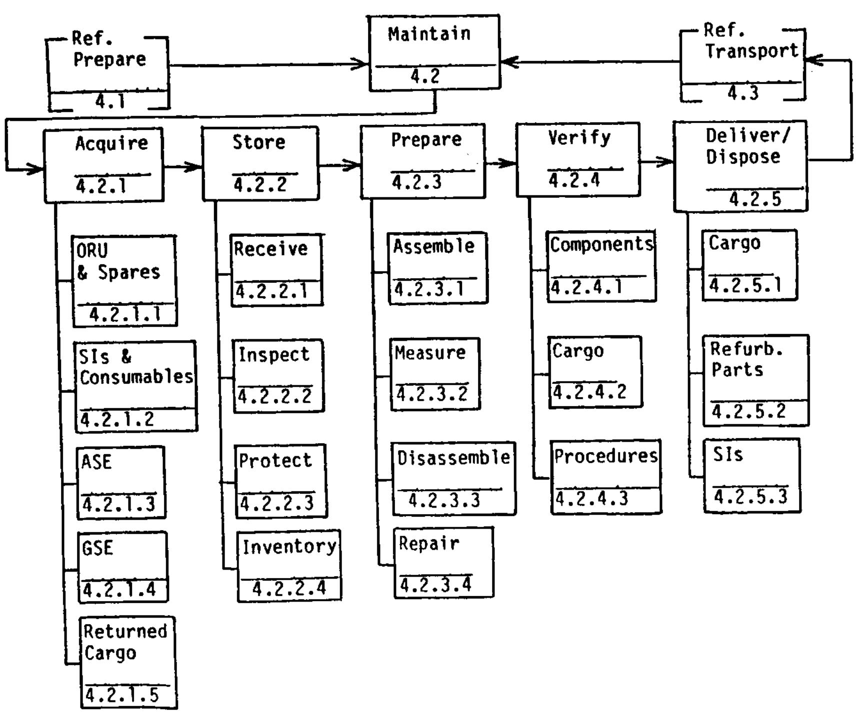

Figure 4 - Satellite Servicing Maintenance Functions

Preparation for use of the Shuttle or Space Station will be standardized through the JSC Payload Integration Plan with its Annexes, but the process for simplification and standardization of servicing preparation tasks for satellite components is yet to be addressed in a similar manner.

To minimize costs and confusion, it is advisable that a standardized methodology be developed and promulgated to all space projects which contemplate being serviced on orbit.

CONCEPT

A standardized servicing operation concept is depicted in Figure 5 for the case in which the Space Station is at a different inclination or is not yet prepared for servicing support. Servicing plans and components are produced for a program in the DDT&E activities. These are brought together for integration in a Satellite Servicing Center. Scientific Instruments and Orbital Replacement Unit spares, with appropriate simulators for neutral buoyancy test, integration tests, and training are received at the logistics and clean room areas. Spares, simulators, panels, tools, and GSE are shipped from the contractor's factory along with a Development/Test Mockup of the satellite, necessary for interface verification, integration and training operations at the center. NASA ASE and its GSE will be introduced when required for verification, integration and training.

The servicing mission planning, integration, training, and cargo assembly are all accomplished at the Servicing Center, from which cargo, ASE and GSE are transported to the launch base. Prelaunch operations at the launch base are kept to a minimum for servicing missions. This conceptual flow is shown in Figure 6.

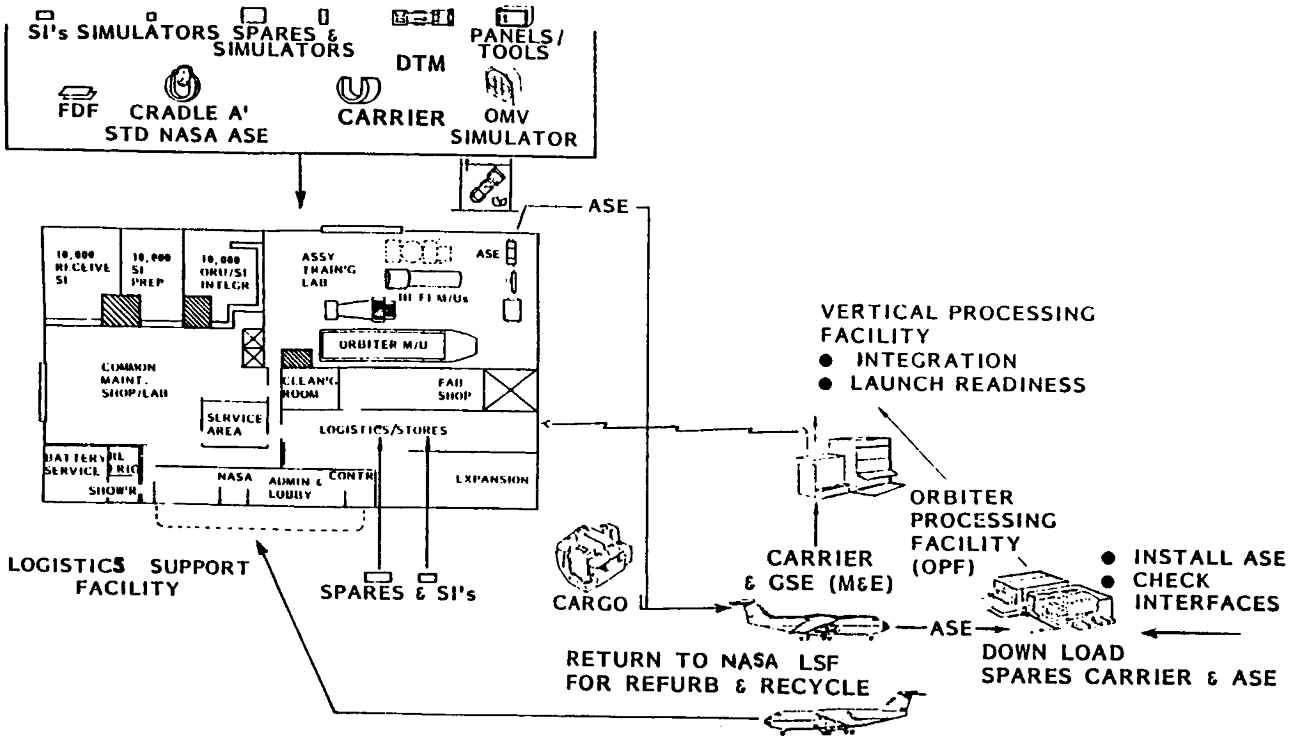

Figure 5 - Servicing Pre-launch Concept

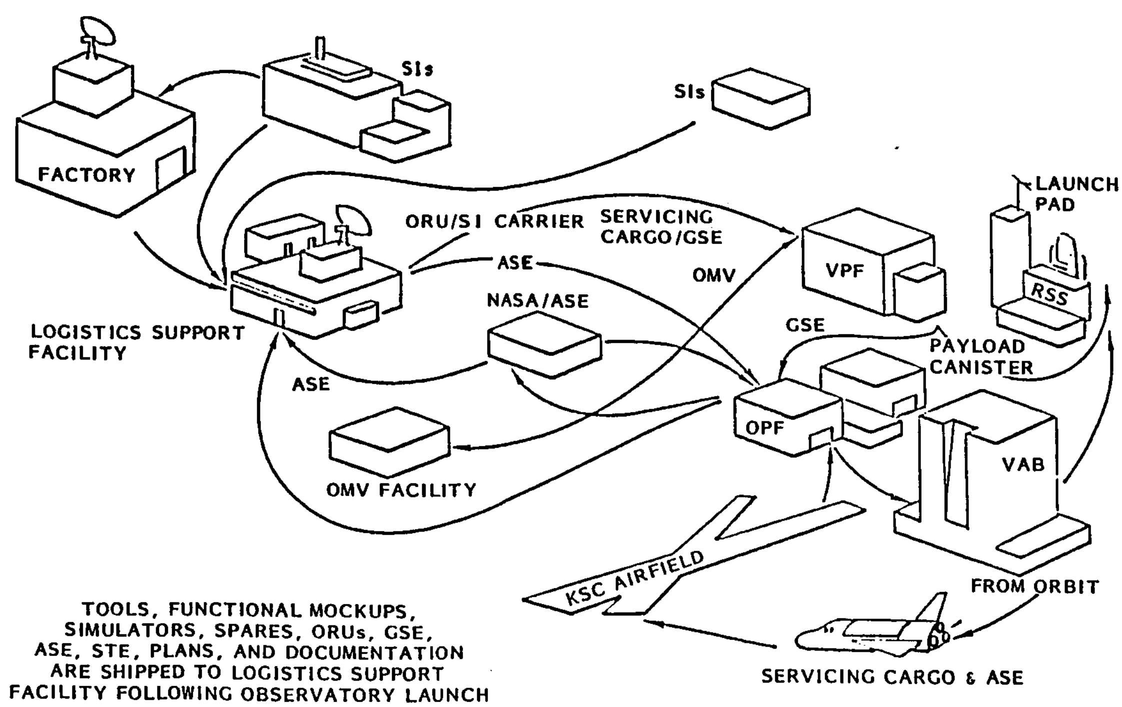

Figure 6 - Servicing Hardware Flow

The Space Station will become the principal on-orbit servicing method for National Astrophysical Observatories and for satellites with the same orbit inclination. Replacement ORUs will be space-lifted to the Station by Shuttle logistics flights, as shown in Figure 7. Servicing free fliers in this era will comprise station-based OMV rendezvous, capture, berthed EVA and robotic replacement and replenishment, and return of changed-out components. The ground logistics, planning, training and verification tasks will be slightly different for Station or Shuttle on-orbit servicing. One difference will be in the lowered complexity of STS integration, since as Shuttle cargo the components will be qualified and then integrated into a standard cargo carrier, to be casual cargo for the orbiter.

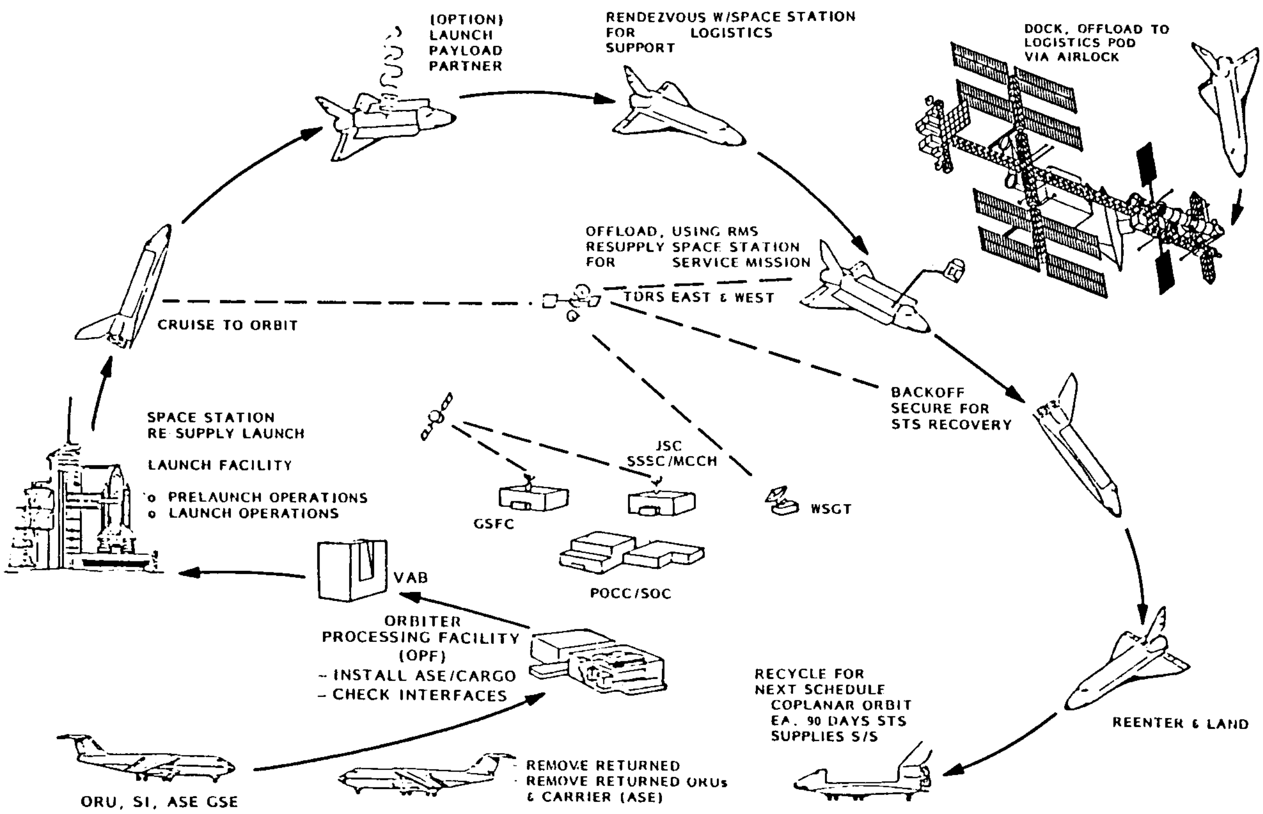

Figure 7 - Shuttle Logistics Flights for Station Servicing

IMPLEMENTATION

Plans and designs for servicing will lead to agency agreement for support and integration activities as shown in Figure 8. During the final year before satellite launch the plans development and interface testing for satellite servicing will grow in level of activity. Training, verification and integration activities will be accomplished to prepare for servicing launch readiness six months after satellite launch. The flight-ready condition awaiting on-demand call-up will be maintained through integrated simulations and training exercises as well as periodic re-verification of critical servicing support elements.

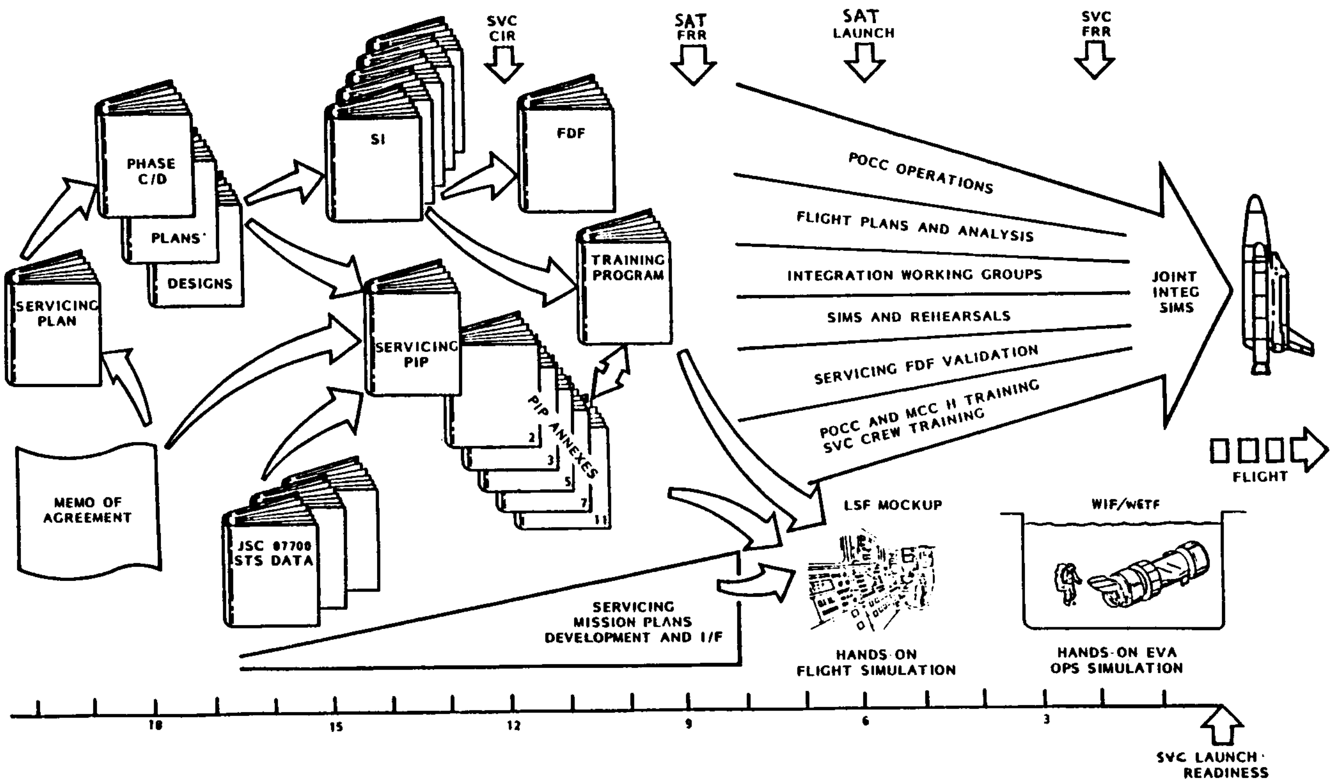

Figure 8 - Pre-planned STS Integration

For servicing, as for other operations, training will be the cornerstone of success. All elements and agencies will be integrated in the training development, and the effectiveness of this integration will be verified by periodic joint integrated simulation exercises.

Servicing Mission Specialists will be trained for the EVA required, while other crew members will receive short-term training based on their assignment to support of a specific flight. These mission specialists will provide project continuity as well as cross-project advances in operations techniques.

Numerous other aspects of pre-mission planning and integration have been explored for upcoming space projects. The logistics of servicing strongly suggests the need for a NASA Servicing Center, sometimes called a Logistics Support Facility, located at whatever site is most prudent. Factors A through G on Figure 9 summarize the results of this exploration.

A. PROJ. SERVICING WILL BE DERIVED FROM OBSERVATORY DEVELOPMENT, AGENCY SERVICES, MISSION OPERATIONS, AND FACILITIES

B. ALL PROGRAM ELEMENTS WILL BE INTEGRATED THROUGH PROJ. SYSTEMS ENGINEERING

C. OBSERVATORY OPS/SERVICING WILL BE CLOSELY COUPLED – DESIGN, DATA BASE, COMMUNICATONS LINK

D. OBSERVATORY EQUIPMENT NOT IN ORBIT WILL BE LOCATED AT SERVICING CENTER DURING MISSION OPERATIONS

E. SERVICING EQUIPMENT DESIGNS, PLANS, LOGISTICS, TRAINING NEED TO BE FOCUSED AT SERVICING CENTER

F. FOR COST AVOIDANCE
   - USE SIMPLE, PROVEN, AVAILABLE SERVICING TECHNIQUES
   - SERVICE ON-DEMAND WITH EFFICIENT LOGISTICS PROGRAM
   - DEVELOP SPACE STATION SERVICING INTERFACES FOR LATER SSP USE
   - SHARE FACILITIES AND TECHNIQUES WITH OTHER OBSERVATORIES

G. FOR RISK REDUCTION
   - DESIGN ALL OBSERVATORY COMPONENTS WITH ON-ORBIT SERVICING OBJECTIVE
   - INTEGRATE ALL SERVICING OPERATIONS AT SERVICING CENTER
   - DEVELOP SERVICING FOR STS WITH SSP COMPATIBILITY
   - VERIFY SI AND ORU COMPONENTS DURING INTEGRATION AT SERVICING CENTER

Figure 9 - Pre-mission Planning Integration for Servicing

After project DDT&E provides servicing planning, equipment, agency support, training plans and facilities, the on-going servicing operations will be conducted at a centralized Servicing Center, called the Logistics Support Facility in Figure 10.

Although this concept has recently been developed for HST and AXAF, it is gaining acceptance as a shared resource for the national astrophysical observatories. Providing this kind of focus of planning and logistics readiness activities is important for cost avoidance and risk reduction.

Servicing mission planning and support will be centered at the Logistics Support Facility which houses the servicing mission planning, logistic activities, pre-flight integration, training and verification operations. The LSF also provides warehousing for spacecraft tooling, GSE and STE that might be needed in the event of a contingency satellite return to Earth, and it provides bonded clean storage for satellite components and spares, SIs and spares, a spares carrier, and ASE for on-orbit servicing.

Since servicing management, engineering, planning, training and operations must continue during the 15-year or greater service life of the satellite, it is imperative that a centralized location for these tasks be established. An on-going logistics data interface is also maintained among the Project Offices, the POCC/SOC, and the various supplier/contractors.

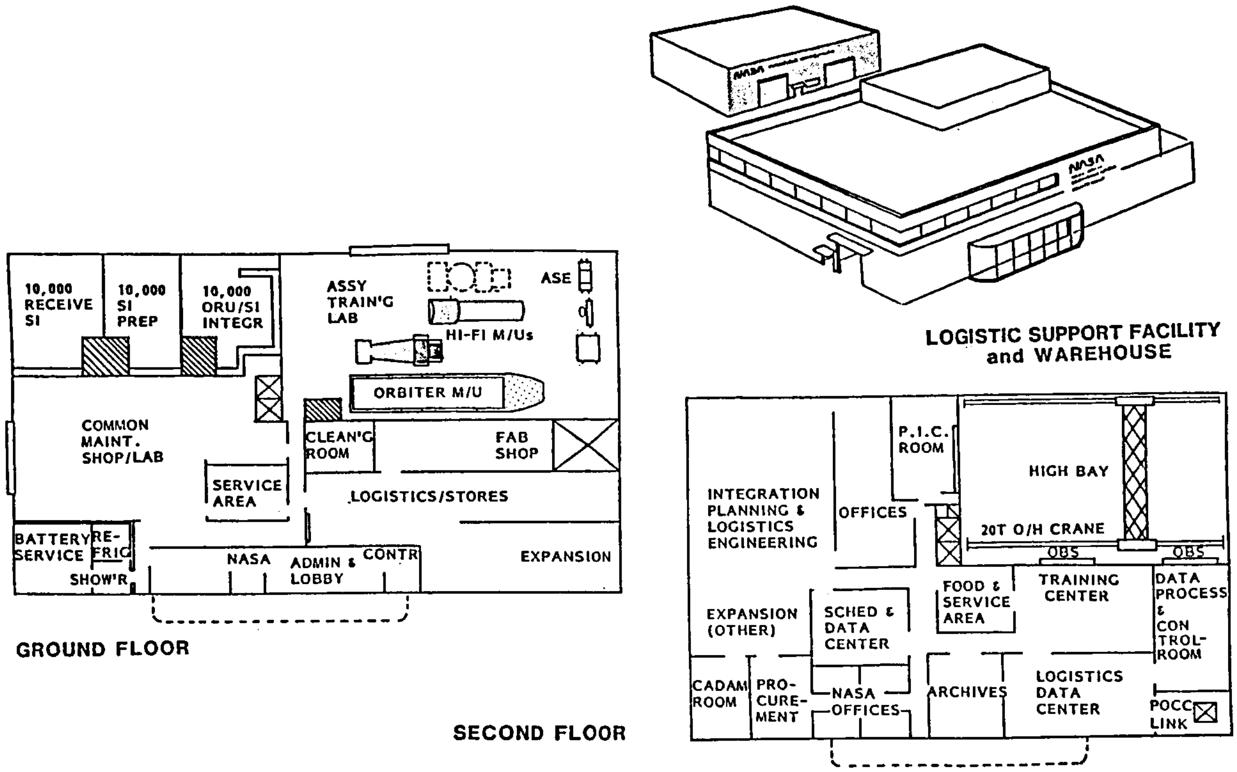

Figure 10 - Satellite Servicing Center

At the time of transition to the Space Station servicing some changes will appear in the Servicing Center. The OMV and berthing systems will be in space, and the Center will have functional mockups in their place. Additionally, in situ control of servicing operations will be from Station crew positions rather than the Shuttle flight deck, so simulation and training will be modified accordingly.

Finally, we must deal with planning and integration documentation, and find a simpler way of producing, updating, controlling and disseminating the information which for some present programs is contained in the project-specific plans and documents shown in Figure 11. The direction for this simplification will have to come from the Government.

STANDARDIZATION

Since the satellite servicing activities will all utilize the same flight systems, whether Shuttle or Space Station, and since launch systems are common national resources, it will be a logical step to develop the national Servicing Center for use by all projects. These facilities will provide a common capability and environment for pre-mission planning and integration for satellite servicing. It follows that the development of this capability will provide the needed impetus for standardization. The facilities being common will generate common pre-mission methodology.

SERVICING OPERATIONS		M & R REPORT SERVICING PLAN		
GROUND OPERATIONS	PLANS	SERVICING PROG. PLAN TEST PLAN QUALITY ASSUR.	TRAINING PLAN SAFETY PLAN FACILITIES PLAN	PAYLOAD INTEG. PLAN LOGISTICS PLAN SOFTWARE DEV. PLAN
	DOCUMENTS	FLT SYST CEI SPEC ASE GSE CEI SPEC LCC ANALYSIS	PRLs ICDs SCHEDULES	
LAUNCH OPERATIONS	PLANS	LAUNCH OPS PLAN LVI PLAN ASE UTILIZATION PLAN PAYLOAD INTEG. PLAN	VERIFICATION PLAN FACILITIES PLAN SOFTWARE DEV. PLAN LAUNCH BASE OPS PLAN	TRAINING PLAN ASE/GSE D&D PLAN SAFETY PLAN
	DOCUMENTS	FLT SYST CEI SPEC ASE GSE CEI SPEC LCC ANALYSIS	PRLs ICDs TIMELINES-ON/OFF LINE	
ON-ORBIT MAINTENANCE	PLANS	SERVICING PLAN PAYLOAD INTEG. PLAN VERIFICATION PLAN	ASE UTILIZATION PLAN ASE/GSE D&D PLAN MISSION OPS PLAN	SAFETY PLAN TRAINING PLAN LOGISTICS PLAN
	DOCUMENTS	FLT SYST CEI SPEC ASE CEI SPEC LCC ANALYSIS	ICDs PRLs TIMELINES-ORBITER/EVA	TRAINING SYLLABUS FLIGHT DATA FILE

Figure 11 - Servicing Operations Documentation

The development of standard methods, specified for use on all projects will be a giant leap toward the reduction of cost, risk and confusion. Integration requirements for functional mockups, simulators, training techniques, logistics interfaces, and even satellite design interfaces will be more easily derived. Test and verification techniques can become standardized, emulating the actual on-orbit servicing to be performed.

As a result of this commonality, then, a standardized set of integration documentation can be developed. The JSC shuttle integration plan, annexes and ICDs might serve as a model. Individual satellite projects would use the basic documents, highlighting only those factors which were unique. Contractors and agencies would welcome this reduction in paperwork, and the products would be more understandable and useful.

SUMMARY AND CONCLUSION

Development of standardized, simplified pre-mission planning and integration is of immense importance to satellite servicing. A national resource for centralized pre-mission activities will enable the spaceflight community to derive common techniques, thus reducing cost and risk to projects which need to include on-orbit servicing to extend life and maximize availability.

Commonality of facilities, equipment, functions, methodology and documentation is an axiomatic goal for satellite servicing.

In developing satellite servicing capabilities, it is imperative that we recognize the significance of preparation for the servicing missions and that we understand why discipline can be achieved only through standardization.